This document consists of 68 pages
Copy No. of 150 copies, Series A

GPO PRICE $ ____

CSFTI PRICE(S) $ ____

Hard copy (HC) 3.00

Microfiche (MF) .75

ff 653 July 65

FACILITY FORM 602

N65-36552 (ACCESSION NUMBER)	1 (THRU)
66 (PAGES)	(CODE)
CR 51831 (NASA CR OR TMX OR AD NUMBER)	31 (CATEGORY)

VOYAGER DESIGN STUDIES.

Volume VI: Development Plan

Avco RAD-TR-63-34

15 October 1963

Prepared under Contract No. NASw-697 by

RESEARCH AND ADVANCED DEVELOPMENT DIVISION
AVCO CORPORATION
Wilmington, Massachusetts
for
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

[redacted]

FOREWORD

The Voyager Design Study final report is divided into six volumes, for convenience in handling. A brief description of the contents of each volume is listed below.

Volume I -- Summary

A completely self-contained synopsis of the entire study.

Volume II -- Scientific Mission Analysis

Mission analysis, evolution of the Voyager program, and science payload.

Volume III -- Systems Analysis

Mission and system tradeoff studies; trajectory analysis; orbit and landing site selection; reliability; sterilization

Volume IV -- Orbiter-Bus System Design

Engineering and design details of the orbiter-bus

Volume V -- Lander System Design

Engineering and design details of the lander.

Volume VI -- Development Plan

Proposed development plan, schedules, costs, problem areas.

[redacted]

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SUMMARY

This report presents the results of a 6-month conceptual design study conducted by Avco Research and Advanced Development Division for the National Aeronautics and Space Administration. The objectives of the study were the synthesis of a conceptual design of an unmanned spacecraft to perform scientific orbiter-lander missions to Mars and Venus during planetary opportunities from 1969 to 1975, and the formulation of a plan delineating the development program leading to first launch during the Mars 1969 opportunity.

The basic approach makes use of a 6000- to 7000-pound orbiter-lander; tradeoff studies were conducted to determine the payload and mission capabilities with smaller and larger spacecraft. The orbiter-lander was selected as yielding the maximum in scientific value short of manned exploration. The lander separates from the orbiter-bus and descends to the planet surface by parachute, where it makes atmospheric and surface measurements and conducts a variety of scientific experiments. The information obtained is relayed to Earth via the orbiter-bus which meanwhile is placed in a planetocentric orbit. The orbiter-bus collects scientific data in transit and maps the planet while in orbit. The lifetime of both orbiter-bus and lander is 6 months for the Mars missions. For Venus, the orbiter life is also 6 months, but the lander life is only 10 to 20 hours because of the hostile environment. A small capsule was designed for Venus, in addition to the lander, to conduct atmospheric measurements after entering from orbit; the capsule does not survive landing. Landers and capsules would be sterilized to avoid contamination of the planets, but the orbiter-bus would be placed on a trajectory which would ensure that it would remain above the sensible atmosphere for at least 50 years; thus, no sterilization would be required. The development plan shows that to obtain the scientific value desired, two spacecraft should be scheduled for each launch opportunity and hardware development should begin in 1964 to meet the 1969 launch date for Mars.

1. INTRODUCTION

The development plan for the Voyager Program is presented in detail herein in three sections. Section I describes the major subsystem and discipline efforts and briefly indicates the Development Program involved. No attempt has been made at this time to expand in detail all facets of each major subsystem development. Identification of the major facilities and special test equipment requirements, the major subcontracted items, the long lead items as well as a list of proposed Government-furnished equipment are included. The major problem areas are briefly discussed. A hardware summary list, indicating the hardware requirements from the development phase to the operational units for each launch window, is also included.

Section II contains the schedules developed for the program. The recommended program awards a hardware contract in the second quarter of fiscal 1965 for a first operational launch during the Mars 1969 launch window. A shortened schedule is presented for this launch opportunity showing program start at the beginning of fiscal 1966. The major penalty for compressing the schedule by 9 months results from a greater overlap between the various development test programs, the qualification test program, and the manufacturing cycle. Modifications necessitated by the test program results will be factored into the vehicle when it is well along in its manufacture. Any extensive modification will become difficult and expensive to accommodate at this point.

Section III contains the cost summaries for the program.

2. DEVELOPMENT PROGRAM

This section discusses the major subsystem development efforts from which the development and hardware costs and schedules were derived.

2.1 Subsystem Development Plan

1. Program Management. Program management includes administrative, engineering, manufacturing, and support services management necessary to ensure effective control of Voyager Program performance and resources. These encompass:

1. Establishment of prime, associate, and subcontractor responsibilities throughout the preliminary design, development, manufacturing, and operational phases

2. Identification of interface requirements

3. Control of the engineering management system which will ensure successful development of systems, subsystems, and components, within stringent reliability and quality control standards, on schedule and within allocated cost

4. Direction and implementation of a manufacturing plan defining tooling requirements, fabrication methods, assembly techniques, production control measures, and engineering liaison responsibilities

5. Direction and implementation of the Test and Evaluation Plan describing in detail proposed quantity, types, and levels of testing to be performed to:

   a. Demonstrate feasibility of the proposed approach to the problem

   b. Demonstrate achievement of performance goals

   c. Demonstrate achievement of reliability goals

6. Direction and implementation of the Reliability Assurance Plan encompassing the details of a continuous program of reliability analysis. This plan will include quality assurance controls delineating procedures by which the contractor will monitor and control reliability and quality activities of the program including subcontracting and suppliers

7. Definition of support services requirements in the areas of finance, contracts, procurement, publications, and other related services.

8. Direction and implementation of a technical and management information service maintaining maximum communication between various program tasks.

a. Performance Control. Program performance is measured and controlled on the basis of established schedules, performance control matrixes, and PERT networks reflecting in detail those tasks as defined in the Development Plan.

Throughout the Development Program, aforementioned performance control matrixes provide engineering management with a master summary of engineering controls and reporting techniques required to ensure timely and adequate performance. Table 1 shows a typical matrix. Task numbers are assigned at the time the requirement for development of a product is identified. These task numbers are used through the accounting system to monitor costs and performance throughout the program. Feedback into the appropriate PERT-Companion Cost network is automatic.

Certain columns under "development test" (equipment, manpower, cost) can be superimposed on any aspect of the Development Program.

Assignments of appropriate control numbers (drawings, specifications, etc., and schedule datesprovide management with a simple, effective tool of central source control for each subsystem and component of the program.

These matrixes are then published through an engineering development handbook which is used as the singular reference (of a comprehensive, yet concise and accurate, picture) of all significant aspects of the program.

Control plans will be developed in areas of Configuration, Interface, Weight and Balance, Manufacturing, Quality Control, Launch Support, Communications, and Documentation.

b. Resources Control. Establish plans reflecting control of:

1) Finance. A measure of dollar expenditure against task to be performed, manpower assigned, material to be purchased, and travel. Financial control will define procedures for cost estimating and funding.

2) Manpower Plan. Definition of numbers and types (skills) of personnel to be assigned to the program phased to development tasks.

3621 INSTRUMENTATION TEST PROGRAM AND SCHEDULE

Task No.	Nomenclature	Drawing Number	Drawing Date	Specification Number	Specification Date	Contract Award Date	Vendor Name	Vendor Purchase Order No.	Delivery to Avco Due	Delivery to Avco Actual	Development Test Test Work Order No.	Development Test Start	Development Test Complete	Development Test Equipment	Development Test Manpower
3621.1	**Telemetry Package Assy.**														
3621.2	**Time Division Multiplexer**														
	a) ACS Monitor														
	b) Commutator 10 x 45 (two-pole)														
	c) Commutator 2 x 45 (two-pole)														
	d) Reference Voltage Source														
	e) Power Control Switch														
	f) Latching Relay Assy.														
	g) Wired Assy.														
	h) Wired Assy.														
	i) Support, Telemetry Pkg. Casting														
3621.3	Dynamic Assy.														
	a) Low-Range Accelerometer														
	b) Medium Range Accelerometer														
	c) Medium Range Accelerometer														
	d) High Range Accelerometer														
	e) Inertial Switch (15-2 g)														
	f) Inertial Switch (0.02 g)														
	g) Low-Range Rate Gyro Assy.														
	h) Medium-Range Rate Gyro Assy.														
	i) High-Range Rate Gyro Assy.														
	j) Rate Gyro Switch Comparator														
	k) Dynamic Assy. Body														
3621.4	Frequency Divider Multiplexer Assy.														
	a) 70 kc + 2.04 SCO (Subcarrier Oscillator)														
	b) 10.5 kc -1 to + 4V SCO														
	c) 14.5 kc -1 to + 4V SCO														
	d) 22 kc -1 to +4V SCO														
	e) 40 kc -1 to +4V SCO														
	f) 0.96 kc 0-4V SCO														
	g) 1.3 kc 0-4V SCO														
	h) 1.7 kc 0 to +4V SCO														
	i) 2.3 kc 0 to + 4V SCO														
	j) 3.0 kc 0 to +4V SCO														
	k) 3.9 kc 0 to +4V SCO														
	l) 5.4 kc 0 to +4V SCO														
	m) 7.35 kc 0 to +4V SCO														
	n) 560 cps ± 2V SCO														
	o) Mixer														
	p) Wired Assy.														
	q) Inverter Monitor														
3621.5	RF Assy.														
	a) Low-Frequency Transmitter (2 watt)														
	b) High-Frequency Transmitter														
	c) Filter														
	d) Power Converter 200V														
	e) Cable Assy. Interface, RF														
	f) Wired Assy. RF														
	g) Cable Assy. RF														
	h) Body Assy.														
	i) Power Amplifier, 10-watt														

TABLE 1

PERFORMANCE CONTROL MATRIX

Qualification Test						Quality Assurance Test Procedure		Acceptance Test Plan		Product Manu-Facturing Order (PMO)		Vehicle Mod							Failure Report		Failure Analysis		Corrective Action		Reliability		
												Usage						First Need									
Cost	Plan Number	Start Date	Complete Date	Results Published (Doc. No.)	Cert. No.	Number	Date	Number	Date	Number	Date	1	2	3	4	5	6	Date	Number	Date	Number	Date	Number	Date	Est.	Actual	

3) Procurement Plan. Definition of make or buy, subcontracter, and material procurement procedures down to the component level. This plan includes identification and qualification of subcontractors and control method of subcontractor progress, costs, and quality.

4) Facilities Plan. Recognition of facility requirements in association with Development Plan tasks including machine loading.

Program management provides task continuity and control from inception of the contract through completion, recording and reporting progress in a systematic manner.

2. Systems Analysis and Integration. System studies will be required to perform overall system tradeoffs to optimize the performance of the spacecraft in attaining the program objectives. The translation of overall program objectives into the various subsystem requirements is accomplished by the systems engineering staff. Systems analyses are required to determine the payload and trajectory parameters for the optimum mission. The development of the mission profile and factory-to-launch sequence of events and the associated, anticipated environmental conditions are keyed to the mission objectives and optimum mission design. Data requirements for the scientific and engineering mission objectives will be established and coordinated.

Interdisciplinary analysis and system synthesis integrating the design effort of the various specialists must be emphasized throughout the program to produce a balanced design accommodating as many as possible of the requirements. Integration of the various subsystems through establishment and maintenance of interface controls will ensure electrical and mechanical compatibility of the various elements within the spacecraft as well as the overall lander-orbiter and spacecraft-booster interfaces. The interface between the Government-furnished scientific instrumentation and the spacecraft data conditioning, communications, power source, and central computer and sequencer equipment will present particularly difficult problems, since the instrumentation design will, in all probability, remain fluid until late in the program. Interface control will be established through coordinated mechanical and electrical interface control drawings and specifications. Mechanical interface control will be maintained through master templates and electrical interface control and testing will require development of electrical simulators for each subsystem involved in the interface. Detailed interface testing will be required to demonstrate the compatibility of each interface.

Participation in the design reviews and all phases of overall systems testing is essential in maintaining cognizance of the system performance and to effect the most meaningful solutions to design problems. Weight analyses are very important in attaining an optimum balance between performance and reliability of the various subsystems. Configuration control is maintained through the systems engineering organization. Close liaison will be maintained with customer and field personnel to prepare for and execute successful flight operations of the Voyager spacecraft.

3. Reliability and quality assurance. In performing unique engineering and scientific tasks, Voyager spacecraft and their highly complex subsystems will be faced with a variety of potential reliability problems. Long term, frequently cycled operation and storage in hostile and poorly defined environments present problems which must be solved to assure a high probability of mission success. The reliability burden is further aggravated by the fact that only few spacecraft will be built to be used during the limited number of favorable launch opportunities. Since the cost per launch is extraordinarily high, it is both technically and economically mandatory that a full-scale, dynamic reliability program be implemented to assure the necessary spacecraft mission reliability.

The relative magnitude of the reliability effort, as compared to the total program effort, is a function of the type of equipment to be delivered. For a maintainable, ground-based radar or aircraft system, a nominal reliability effort, funded at 5 to 10 percent of the total program costs, would be sufficient since repairs and preventive maintenance could be employed to preserve their operational readiness. However, the preservation of operational reliability through maintenance is not relevant in the case of the nonrepairable interplanetary probe, Voyager. Furthermore, since the reliability of the booster is a severe constraint to the success of the mission, the reliability of the Voyager spacecraft must be necessarily higher. To assure this high level of spacecraft reliability, a dynamic reliability program is essential.

The primary emphasis in the Voyager program will be on designing high reliability into the spacecraft and maintaining this level of reliability during the production phase. Since a limited number of spacecraft will be built, their fabrication will be more associated with a model shop operation than a high-volume production facility. Meticulous attention to detail will be an overriding concern throughout. Reliability progress will be monitored in vendor/subcontractor liaison and a continuing effort maintained throughout the program.

An ambitious reliability program for subcontractors and vendors must be imposed to assure that reliability objectives are not compromised. Subcontractor and vendor reliability efforts will be funded separately on reliability contracts or reliability requirement clauses. These funds will be apportioned out of the reliability program funds. Separate reliability contracts or clauses will not only mean specifying reliability requirements, but also providing a rigorous monitoring effort to enforce these requirements through incentives and penalties. The monitoring aspects of the program will be provided through a carefully conducted audit and surveillance of the subcontractor/vendor efforts, resident reliability engineers, as well as requirements for reports submission.

The technical function will involve such important activities as reliability engineering, test engineering, and data functions. The reliability engineering activity will be responsible for subsystem reliability allocations, design reviews, and reliability appraisals. The test engineering activity will be responsible for the evaluation of all test programs, both in-house and in the field. This responsibility will include test planning, test monitoring, and laboratory analysis of failures. Data functions will be conducted for both reliability engineering and test engineering activities. A particularly important data activity will be that of tabulating and reducing data reported from the various test programs. These data will then be analyzed by the reliability engineering activity to initiate reliability improvements, thus completing the corrective action loop. The above technical functions will be maintained throughout the Voyager Program. The complete reliability development plan is described in table 2.

The reliability assurance philosophy recommended for the Voyager Program is founded upon three essential precepts.

a. Direct and unimpeded access to top management. The necessity of an organizational structure which will allow an unrestricted management information and control system for the timely recognition and effective correction of reliability problems.

b. Meticulous attention to detail. The necessary policies and procedures and the careful selection of qualified personnel to anticipate, detect, rectify, and follow up existing and potential reliability problems in the system design and operational hardware such that even the most minute aspects are given adequate attention.

c. Exhaustive testing. The thorough evaluation in all phases of the program to ascertain that the system hardware in its functional operation has the inherent ability to perform as intendedin the anticipated environments and the necessary corrective action to remedy all deficiencies.

The translation of this philosophy into certain essential elements of control and the implementation of these controls will assure the operational reliability required in the program. These controls provide for adequate planning, influence of the design, complete testing evaluation, quality manufacture of system hardware, fastidious launch site preparation of operational equipment, and, above all, a dynamic means to achieve reliability improvements through early problem identification and meaningful corrective action.

Volume 3, Systems Analysis, contains the results of the reliability analysis conducted during this study, and shows the proposed level of testing which would be required to demonstrate the achievement of the reliability goals assigned to each subsystem. This analysis would be carried out in greater detail and continued for the duration of the program to obtain the most realistic reliability goals, and also to record progress in achieving these goals. The design of each subsystem would be reviewed on a continuing basis, and the results of development, reliability, and environmental testing would be reviewed at all stages. Reliability would recommend design changes, such as the incorporation of redundancy, to meet the desired reliability goals.

LEGEND

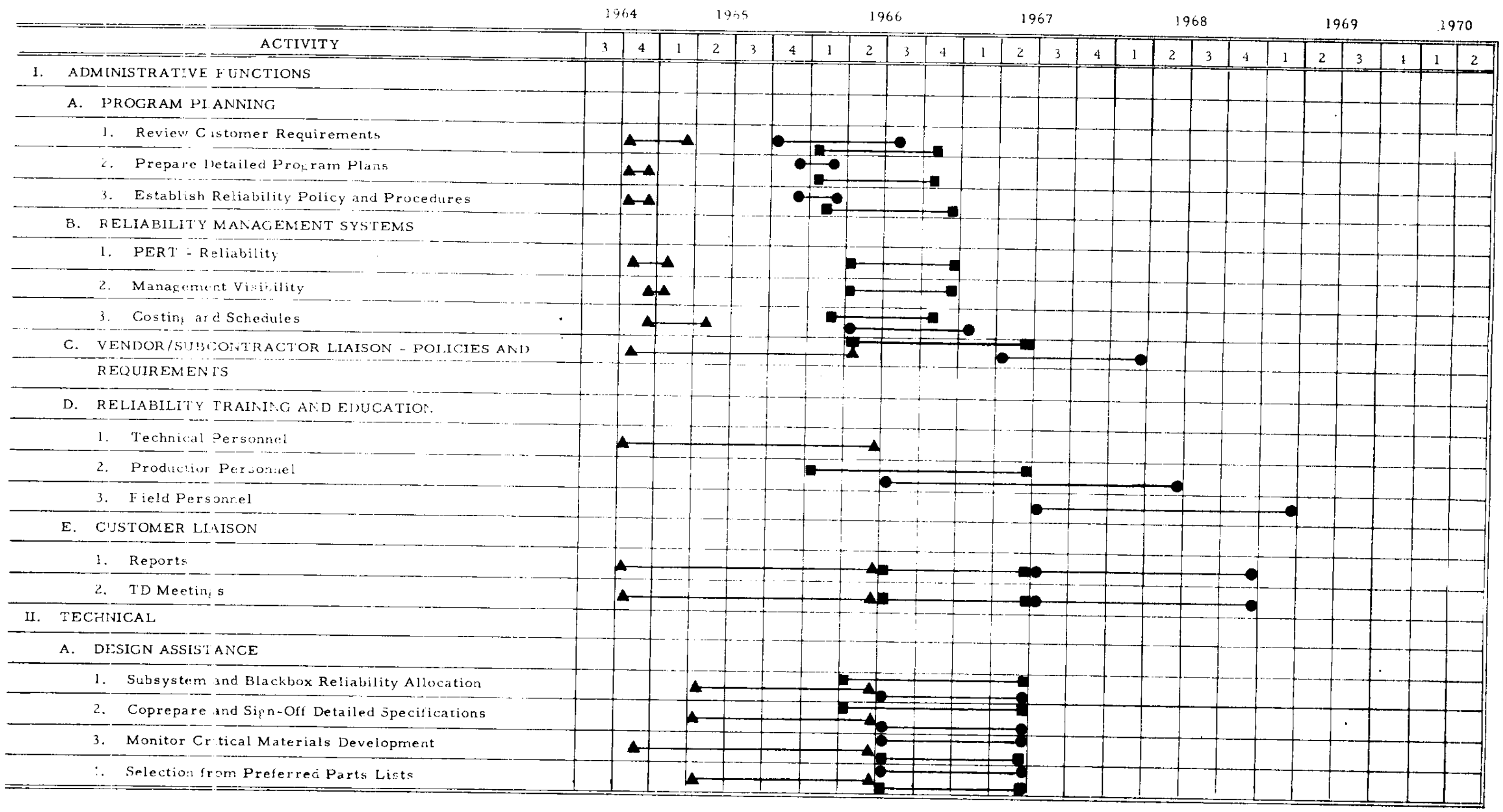
ENGINEERING MODEL PHASE

PROTOTYPE PHASE

PRODUCTION PHASE

TABLE 2

RELIABILITY DEVELOPMENT PLAN



LEGEND

ENGINEERING MODEL PHASE

PROTOTYPE PHASE

PRODUCTION PHASE

TABLE 2 (Cont'd)

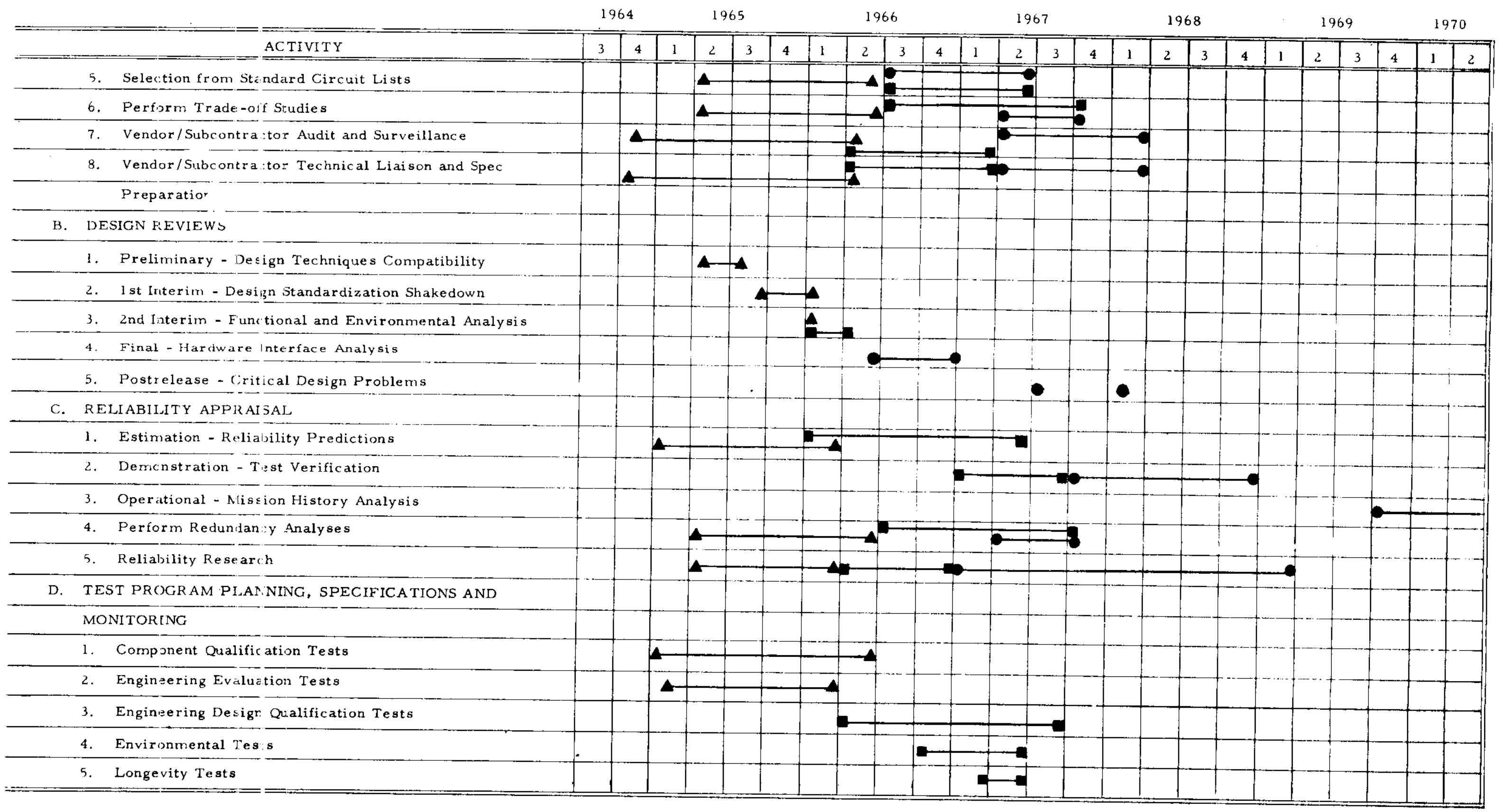
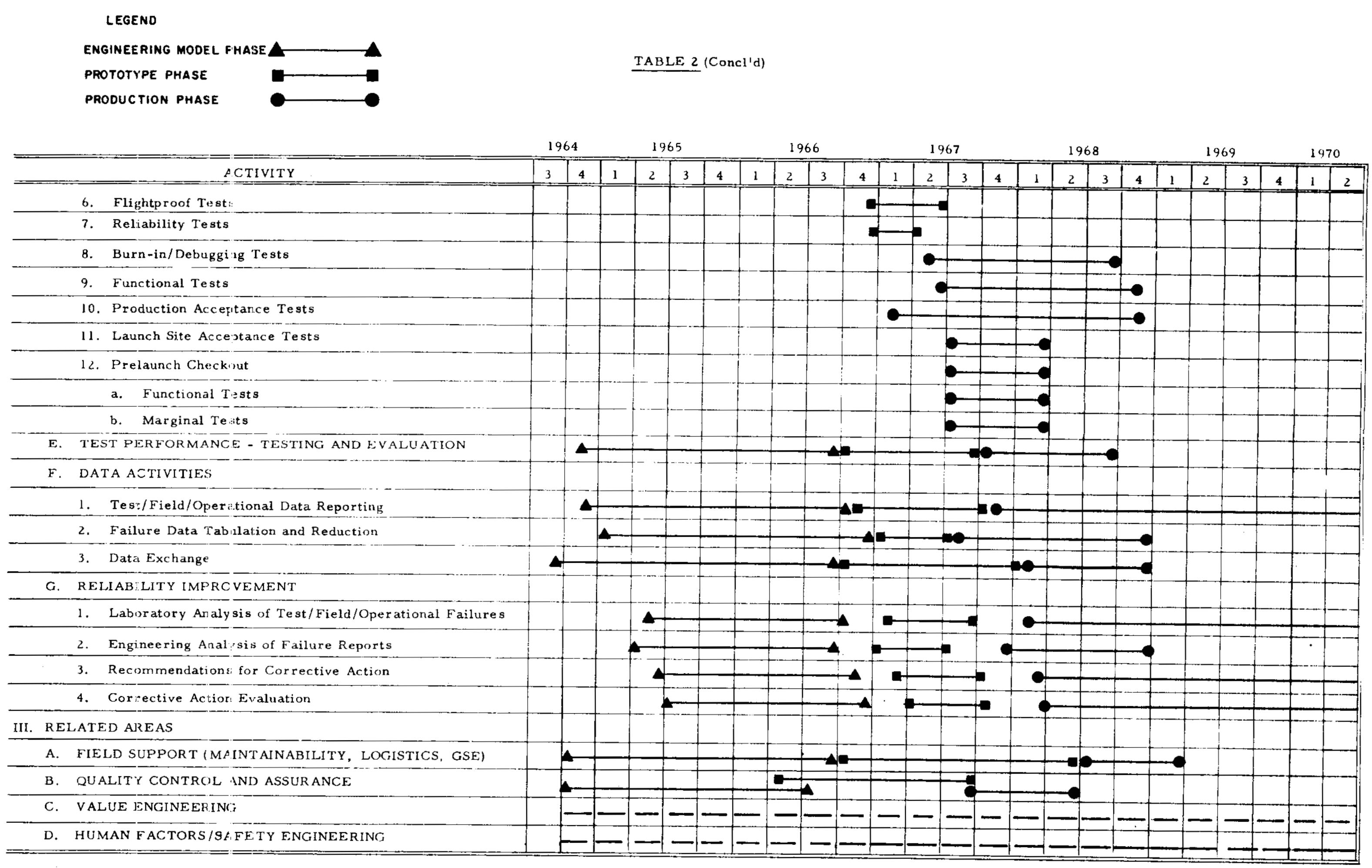


TABLE 2 (Concl'd)



4. Spacecraft design. The spacecraft design effort embodies the mechanical and electrical integration of all spacecraft subsystems into a coherent overall design for the Voyager spacecraft. Detail design considerations will involve the booster-spacecraft and orbiter-lander separation system requirements and the lander erection and electromechanical deployment system. The effort necessary for preparing the complete design definition of the spacecraft requires layout drawings, detail production drawings, and subsystem component and process specifications. Mockups and full-scale models will be built to facilitate internal packaging design. Prototype or development models of each spacecraft configuration are required to ensure proper operation of each subsystem, and to facilitate integration of the science payload.

A detailed Development Program will be required for materials, components, and subsystems of the electromechanical devices outlined above. Factory and field liaison will be provided during the manufacturing, assembly, acceptance test, sterilization, and operational portions of the program.

5. Structural development. The structural development program consists of an analytical and structural test effort required to perform a design optimization study of the orbiter-bus and lander structure and to substantiate the structural integrity of the final design. During the initial design phase, preliminary aerodynamic and thermodynamic environments are used in conjuction with design and environmental criteria dictated by the ascent, space flight, and planetary entry conditions to obtain the applied load history to be sustained by the structure. Stress analyses are performed to optimize the selection of materials and element sizes. Complementing the analytical effort, a structural development test program is conducted in which any complex structural conducted to substantiate the applicability of the optimized structure to the fabrication techniques proposed.

During the final design phase, an extensive detailed analytical effort is required to design a minimum weight structure considering the detailed aerothermoelastic problems. Extensive full-scale testing is necessitated by the difficulty in projecting the structural behavior of the spacecraft under complex loading conditions. Tests and detailed analysis will include the time sequencing of the ascent, space flight, and entry environments. Test facilities for simultaneous simulation of the aerothermoelastic environments do not exist in the United States for vehicles of this size and weight. Development of non-existent test facilities will take place during the spacecraft design development phase so that the full-scale structural proof tests can be combined with the detailed analytical effort to provide the assurance of a structurally sound final design.

6. Aerodynamics development. To establish the optimum aerodynamic configuration that will fullfill the overall system objectives, it is necessary to conduct studies to determine vehicle trajectories, vehicle force and moment coefficients, static and dynamic stability, pressure distributions, convective

and radiative heating, and inertial and air loads as well as vehicle shape. The governing parameters in the above studies are associated with the planetary atmospheres. A vehicle configuration and design of high confidence and reliability will require extensive analyses investigating the influence of all possible atmospheric models. These studies will be modified as data on the planetary atmospheres become available from such sources as Mariner B data.

The overall systems design influences the selection of aerodynamic configurations for these particular vehicles. Most significant of these are (a) maximum electronics package weight, (b) reliable descent and impact attenuation system weight and performance, and (c) static and dynamic stability.

For detailed design analysis requirements, a complete definition of aerodynamic coefficients as a function of Mach number and angle of attack will be obtained as well as the variation with gas composition. This is necessary to ensure adequate static and dynamic stability as well as to determine the required design conditions from trajectory analyses.

Entry trajectory analyses will be performed to obtain the necessary environmental conditions. Three-degree-of-freedom trajectories will be obtained to establish the altitude-descent histories for the various atmospheres to be considered. A dynamic analysis is necessary to establish the stability and angle of attack envelope. The necessity of a six-degree-of-freedom analysis will be investigated and performed if necessary to obtain detailed variations in dynamic pressure and altitude histories. These analyses will establish the design criteria for determining loads, heating, and angle of attack effects. The criteria for a reliable descent system will be obtained and, in addition, error analyses will be performed to establish the descent system design.

Pressure distributions and loads will be initially evaluated by the method of characteristics utilizing effective ratios of specific heats to account for the variations due to gas composition. This and other assumptions in the preliminary design phase must be corroborated by test programs.

The heating evaluation will be done by existing techniques and correlations supplemented by test data as they become available. The trajectory analyses, especially the dynamic angle of attack effects, will be used to ascertain the design heating pulse, both radiative and convective. Flow-field analysis will be required to establish the heating effects at angle of attack and to determine the pressure distributions and loads.

A wind tunnel test program will markedly influence the determination of the aerodynamic coefficients. The subsonic test program can be carried out in any of the continuous separated wind tunnels such as AEDC tunnel A or the JPL 20-inch facility. Testing at Mach numbers from 6 to 10 requires use of the AEDC tunnels B and C or the JPL facility. The hypersonic test program will make use of the hot shot tunnel at AEDC or the 48-inch shock tunnel at the

Cornell Aero Labs. As wind tunnel test data become available, the results will be factored into the design.

7. Heat shield development. The heat shield development program will involve design, test, and evaluation of the Mars and Venus heat shields. Preliminary thermal design evaluations will be conducted to select a suitable heat shield material and to size the lander heat shield and structure. The initial design effort generally utilizes preliminary environmental estimates and material properties. Further refinements in the aerodynamic heating and material property behavior under simulated Mars and Venus entry environments will lead to an increase in heat shield design confidence. A near exact analytical ablation model will be used together with computer approximation techniques to arrive at the required solutions. Material property inputs required for the analytical models will come from a series of basic tests. These tests include (a) convective heat flux tests, (b) radiant heat flux tests, (c) combined convective-radiant heat flux tests, (d) solar furnace tests, (e) thermal conductivity, thermal expansion, and specific heat tests, and (f) thermogravimetric analysis (TGA) and differential thermal analysis (DTA). Candidate materials will be screened in both convective and radiant energy arc facilities, using the Apollo ablator as the reference material. Selected materials will then be evaluated under simulated Mars and Venus entry conditions, using all suitable heating facilities, including the combined convective-radiant heat flux arcs. Thermal properties and densities of these candidate materials will be obtained. The kinetics and mode of degradation will be determined with TGA and DTA tests.

The following series of major mechanical and chemical tests will be performed (a) tensile and compressive strength, modulus, and elongation versus temperature and strain rate, (b) bond strength versus temperature and test rate, and (c) chemical analysis of candidate materials and their products of degradation. Basic tensile and bond properties will be obtained at selected temperatures and strain rates for all candidate materials. Selected materials will be completely evaluated for mechanical and chemical properties. Test conditions will be determined by ground, launch, in-flight, and entry conditions. These properties will be engineering data upon which a design freeze can be based.

After the prescreening test, environmental tests will be conducted on candidate materials. These tests will include (a) high-vacuum exposure ($<10^{-5}$ torr), (b) temperature cycles, including cold soak, (c) solar irradiation, (d) combined high-vacuum and solar irradiation, (e) heat soak-sterilization conditions, (f) effect of humidity and thermal aging, and (g) shock and vibration.

The effect of these environmental conditions will be evaluated by thermal, ablative, mechanical, and chemical tests. Large-scale testing will be conducted on panels and mockups by evaluating structural integrity, weight change, stress and strain levels, and visual changes in the heat shield.

8. Thermal control subsystem development. Initial systems studies will establish the type and extent of thermal control required for the Voyager orbiter and lander for both Mars and Venus missions. The initial study will consider both active and passive thermal control. Both external and internal variable thermal loads will be considered in the preliminary and detail designs. The recommended thermal control system will be developed and evaluated analytically through digital computation techniques considering mission reliability requirements as design objectives. Initial subsystem developmental testing will be performed to establish general design criteria, such as coating degradation, component thermal dissipation and load paths, duty cycle transients, and mission midcourse maneuver effects.

Substantiation of analytical results will be accomplished through experimental thermal vacuum testing under simulated flight conditions with the appropriate complement of subsystems in operation. Detail problem areas will require further investigation with possible redesign as a result of the thermal test program. A continuing review of thermal capabilities of all subsystems will ensure the successful operation of the Voyager spacecraft and lander under widely divergent conditions experienced during exploration of the two target planets.

9. Communication subsystem development. A preliminary design effort will be conducted to determine the requirements of the communications system considering the demands of the desired scientific mission, the available weight, the required operation life, thermal environment power available, and the sterilization constraint. From the system requirements, detailed specifications will be generated for all subsystems.

A supplier will be selected for each of the subsystems based on experience with similar equipment for long-duration, high-reliability space programs, conformance to the program schedule requirements, cost, and reporting practice. During the subsystem design period, an intimate relationship will be maintained within the subcontractor to ensure proper exposure of problem areas and schedule slippages.

Prototype models of all subcontracted items will be subjected to a series of development tests both at the vendor site and at the prime contractors facility. These tests will include (a) bench testing to determine the conformance to the electrical specifications, (b) RFI testing, and (c) design data environmental testing.

Modifications will be incorporated in any areas where the test item fails to meet the specification requirement. Upon completion of the subsystem development cycle, a communication system prototype will be fabricated and subjected to the electrical bench testing, environmental testing, and RFI testing. A prototype of the data conditioning system will be constructed and utilized as a compatibility mockup for scientific instruments.

During the manufacturing and assembly phase, engineering personnel will be assigned to act in a consultory capacity for inspection, assembly, and test personnel. In addition, liaison will be maintained with field personnel to allow rapid resolution of problems arising in the communications system at the launch site.

10. Power sources development. An RTG has been selected as the power source for the Mars lander. The power presently considered to offer the optimum match between weight, mission effectiveness, and thermal limitations is 110 watts.

A fundamental consideration with an RTG is that of isotope selection. Because of the long lead time involved, an early request must be made to NASA for AEC commitment of a sufficient quantity of the Pu-238 fuel. Also, early in the program (within 3 months), the physical configuration of the generator will be established by thermal and spacecraft constraints, and the subcontractor will be selected for development of the thermoelectric elements.

Fuel safety criteria, a shielding analysis, ground-handling techniques, and test procedures will be established and, since the capsule will be of the intact reentry design, modes of entry, impact and the resulting stresses will be specified so that a suitable fuel capsule may be developed. Fuel capsule tests will include impact, fire, explosion, and simulated entry.

Throughout the entire program, close liaison will be maintained with the subcontractor. Life testing of the selected conversion material under simulated condition will require 1 full year. This will not significantly retard the development schedule as it serves only to establish the necessary confidence in the material. Concurrently, other elements will be used in the thermal mockup of the generator. Final generator design will follow.

It is expected that the fuel will be delivered about 2 years after commitment (which will be about 18 months before launch). This will allow sufficient time to complete functional, environmental, life, and quality testing such that the final flight proof article is available at least 1 year before launch.

The primary power source for both the Mars and Venus orbiters consists of solar panels with batteries as the secondary source. Development models of the entire solar panel will be constructed. The resulting panels will be submitted to extensive laboratory testing to ensure conformance to the requirements of the detailed specifications and performance criteria. Thermal vacuum tests will be conducted to establish the adequacy of the solar panel design in supplying the power needs of the orbiter vehicles.

11. Mapping subsystem development. During the preliminary design phase of the Development Program, a thorough analysis of the mission concept for both Mars and Venus launches will be conducted, including studies to determine the orbits achievable for all opportunities, the communications capability available,

and the exact scientific objectives desired. From these studies the requirements for an optical mapping system and a microwave mapping system will be generated. Subcontractors with demonstrated capability in the subject fields will be selected by competitive bidding. During the development phase, extensive testing of the mapping equipments will be conducted at the subcontract agency, at the prime contractor's facility, and possibly flyby tests at a field site. The laboratory testing will include thorough examination of the electrical and optical performance of the television mapping system, complete electrical performance testing of the microwave mapping system including antenna pattern measurements, and environmental testing of all mapping subsystems.

During the fabrication assembly and testing phases, close liaison will be maintained between the design engineers and manufacturing personnel to eliminate problems associated with improper understanding of the mechanisms of operation of the systems.

12. Guidance and stabilization control system development. The development of the guidance and control systems will center on two predominant areas. The first is the long mission life, and the effect this has on reliability. The components used for the system must undergo rigorous environmental and life testing to ensure that they will operate for the time required. This imposes the most critical demands on the gyros, the cold gas reaction controls, and the computers. Although the gyros selected are air-bearing gyros, and are expected to have the reliability and lifetime required without major redesign, the desired level of confidence in this performance will require testing. The selection of a cold gas reaction system will require the least amount of development work because of the advanced state of development of these systems. Initially, a parallel development of an alternate technique will be pursued, either hypergolic hot gas or sublimating low-pressure jets, to ensure a backup capability if later knowledge of micrometeroid and other torque-producing effects results in excessive growth in the weight of the cold gas system. The computers in both lander and orbiter-bus present a reliability problem simply because of their large numbers of components. Existing computers do not meet the total requirements of the system, and must be developed, although it is possible that existing computers can be adapted to the needs of the orbiter-bus. The computer in each case will have to be developed to handle the large number of programming and sequencing functions.

In addition to the development required to achieve reliability with existing or nearly existing components, entirely new components are required to meet performance requirements. This is particularly true of two of the on-board optical sensors, the planet tracker, and the horizon scanner. The planet tracker is required to determine range from the planet and represents a major component development. Since many of its features are similar to those of the optical mapping system, these efforts will be conducted in close collaboration. The possibility of a single instrument meeting both requirements will be vigorously pursued. The horizon sensor is an important development area not only because of its effect on accuracy of on-board position measurements, but also because of its use in positioning the planet-oriented mapping instruments. The existing sensors for Earth are not directly suited for Mars, and those in development for Mars may not achieve the accuracy and lifetime required for Voyager.

Detailed system analyses will be performed to determine performance requirements of all components. These results will be modified and integrated to meet total systems requirements, and will be used to develop detailed component specifications and subsystem performance specifications.

Subcontractors will be selected for each of the major subsystems or components to perform the development and manufacture. Component testing to detailed specifications will be performed by the subcontractor. Engineering models will be delivered to the prime contractor for incorporation in subsystem and system testing. Subcontractors will perform reliability, quality assurance, and acceptance testing on their components. These tests will be repeated on the subsystem level at the prime contractor's facility.

Single-axis testing of the reaction-control system will be conducted using a one-degree-of-freedom table and complete system simulation will be performed on a three-axis flight simulator connected to an analog computer. The three-axis, air-bearing table will be used for final system tests, and to simulate the long mission endurance requirements of the control system. This will be combined with a celestial simulator so that closed-loop performance of the optical sensors can be coupled with the reaction control system.

Particular emphasis will be given to the analysis of results from Mariner flights. If the results indicate attitude control system demands greater than those estimated due to solar pressure, micrometeroids, and other effects, the emphasis on alternate techniques will be increased.

Close liaison will be maintained with the propulsion subcontractor, since the attitude control system used for thrust vector control is an integral part of the propulsion system, but is controlled by the guidance and control system electronics and logic.

13. Propulsion subsystem development. The Voyager propulsion program will entail the design, development, qualification, and delivery of a Voyager orbiter propulsion system, and a Mars lander, and Venus capsule propulsion system.

The first 2 months of the design will include a review of all commercially available hardware prior to starting component fabrication and/or procurement.

The first 6 months will be utilized in analysis and generating component design drawings, manufacturing design drawings, laboratory test specifications, fabrication procedures, and vendor specifications. Reliability statistical testing will be stressed throughout the program. Laboratory and injector hot-firing tests for Voyager component system development will commence during the latter half of the first year. Component testing will include a separate test program for each major component to determine reliability and possible performance gains. Specific tests will be conducted on thrust chambers, valves, and injectors to determine operating characteristics and for performance evaluation. Tank diaphragm testing will be conducted for verification of expulsion efficiency. Nearly all of the component testing will be accomplished at the propulsion subcontractor's facilities. Testing at vendor facilities (chosen by the propulsion subcontractor) will be necessary to accomplish all of of the environmental, vibration, burst, acoustical, and vacuum requirements.

Hot firings will be conducted at the propulsion subcontractor's field laboratory. Initial system testing will be conducted with "work-horse" hardware at ambient conditions. Prototype system testing will follow the initial test and will include altitude, acceleration, shock, and vibration testing. These tests will encompass the extremes of system operating limits to verify satisfactory operating limits to verify satisfactory operation in excess of specification requirements.

Prequalification system testing in scheduled to start approximately 25 months after go-ahead. The qualification program will be initiated almost at the same time of the prequalification testing and will probably continue from 2 to 3 months after the prequalification program has been completed.

14. Descent and reerection system development. System analysis will be conducted to determine the characteristic performance required from the descent and reerection system. The system design will attempt to encompass as large as possible a variation in the Martian atmospheric model. However, where necessary, emphasis will be placed on the model of current highest probability. Selection of the descent system subcontractor will be based on competitive bidding. Only contractors with demonstrated capability in this technology will be considered. Close liaison will be maintained with the subcontractor during the entire development program. Extensive functional and environmental testing will be necessary to evaluate and demonstrate the performance of the descent and reerection system. Tower and aircraft drop test programs described elsewhere will be a major factor in the evaluation of this subsystem. Considerable factory and field liaison will be required during the entire manufacture, qualification test, assembly acceptance test, and operational phases of the program.

15. Aerospace ground equipment development. The aerospace ground equipment, in support of the spacecraft, will consist of electrical checkout equipment and mechanical handling, transfer, and shipping equipment for the assembled spacecraft and its subsystems and components. During the spacecraft development phase, AGE support will maintain close liaison with all areas of the design effort. As the specifications and performance criteria are developed for the spacecraft and its subsystems and assemblies, the requirements for the AGE support equipment are identified and formulated and specifications

are evolved. The design is then tested for its compatibility with the various mockups and developmental units of the orbiter, lander, a complete spacecraft, subsystems, and assemblies.

Included in the subsystem tests performed by the electrical AGE are the following:

a. During the qualification test program, functional checks will be performed on spacecraft systems to demonstrate capability of equipment to operate under required environmental conditions.

b. Prior to assembly into spacecraft, checks will be performed on each subsystem. Equipment will be provided as required to simulate interfaces during subsystems checks.

c. System checks will be performed on complete spacecraft in the factory prior to disassembly for shipment to launch site.

d. Functional checks will be performed at the field test site on spacecraft subsystems upon receipt from factory.

e. System checks will be performed on assembled spacecraft at the field test site prior to mating to a Saturn booster.

f. Checks will be performed on a Saturn booster prior to spacecraft mating to assure compatibility between booster and spacecraft.

g. Launch site checks will be performed on mated spacecraft to provide assurance that the system is operating properly prior to launch. The electrical ground equipment required to support the spacecraft will be incorporated into the Voyager system test set to perform both subsystem and system level checks on the instrumentation, guidance and control, communications, scientific instrumentation, and power supply subsystems of the orbiter. It will also perform checks on the navigation, communications, power supply, and scientific instrumentation subsystems of the lander.

Major facility requirements for the performance of the electrical checkout include air-bearing tables and astrosimulators. These facilities are required at each test site to physically support the spacecraft during closed-loop dynamic performance checks on the attitude control system. The table will be installed in an astrosimulator which will simulate the positions and brightness of the sun, Canopus, Mars, and Earth during the checks.

The mechanical handling, transfer, and shipping equipment will be designed and developed in parallel with and for use during the spacecraft development. Although most of the equipment is similar to that used on space and reentry vehicle programs, some difficult areas exist. Among these are (a) the shipment of an assembled orbiter, (b) handling and shipment of the RTG unit, and (c) handling and shipment of the pyrotechnics peculiar to the program.

16. Sterilization program. A sterilization facility will be required to sterilize and certify the sterility of lander flight hardware. It will be necessary to design, construct, equip, staff, and check out this facility. A pilot sterilization plant, a working model of the actual facility, will be constructed to establish and evaluate the sterilization approach which is to be used. This facility will allow acquisition of statistical data upon which the sterilization procedures can be based. Two formal sterilization facilities will be required for this program. A complete facility at the lander assembly site will allow initial sterilization of lander hardware. A second complete facility should be erected at the launch site; however, the second facility can be reduced to include only the terminal portion of the facility at small risk to the program. An extensive training program for sterilization personnel must be established to adequately train the assembly and certification technicians. The sterilization facility must then be checked out using mockups and development hardware of the final configuration to establish the adequacy of the sterilization procedures. The sterilization facility must be rechecked at regular intervals to ensure that no relaxation of the rigid controls has occurred. A more detailed description of the sterilization facility is presented in volume 3.

17. Test support program. The flight planning, launch support, and performance evaluation activities for the Voyager program include (a) preflight planning and support of the range requirements, (b) field activities, and (c) postflight engineering evaluation.

General mission planning is directed toward development of the overall test plan and method of accomplishment. Consideration of the program objectives, design configuration, and launch constraints will provide the basis for determining the necessity for flight tests, ground tests, and engineering data acquisition.

A ground-support system plan will be developed and documented early in the program. This will integrate the tracking, communications, data acquisition, and range support requirements necessary to support the interplanetary launch. Final definition of the requirements will necessarily be a cooperative effort with NASA, JPL, range personnel, and participating contractors.

Early in the program, an orbiter-lander handling and test philosophy is developed which covers the factory through launch phase. This philosophy is based on (a) the analysis of each system and its individual checkout requirements; (b) the overall orbiter-lander configurations and the restrictions or special requirements which it imposes on handling devices and techniques; (c) the requirement for clean room operations and lander sterilization; and (d) consideration of the test-station range-safety requirements. As the program progresses, the other ground test planning activities are being accomplished and their results fed back in support of the original concepts. Based on these data, assembly and test procedures will be prepared and published. These procedures are provided to the field for their use in processing the orbiter-lander.

As a means of verifying the handling and test philosophy and the compatibility of the support equipment, both the orbiter-lander and itself, a compatibility test exercise will be conducted. This exercise will occur as early in the program as possible, but is contingent on the delivery of the first of each type of support equipment to be fabricated and the availability of a model orbiter-lander for use in the exercise. Because of these contingencies, an exercise such as this normally cannot occur until just before implementation of the program at the field station.

The exercise will be conducted at the home plant so that the disclosure of any discrepancies may be speedily resolved. The operations to be performed and checked will be identical to those documented in the assembly and test procedures. The exercise will also serve a secondary purpose, that of providing a means of familiarizing the field personnel with the equipment and procedures.

Analysis will be made of the environments to be encountered, the system functions to be performed, and the data acquisition capabilities from launch through planetary encounter. From this analysis, requirements will be established for the engineering measurements to be made aboard the orbiter and lander. The final output of this task will be a formal data specification for the orbiter-lander listing measurements to be made, range, sampling rate, and justification for these measurements.

The orbiter-lander systems and methods of testing will be evaluated both from a standpoint of safety to personnel and gathering of data for submission of a range safety package. Special consideration is given to those systems and circuits which contain pyrotechnic devices or radioactive material. The evaluation of these systems concentrates on proper safing techniques, proper use of test equipment to check them, and exercising the necessary precautions for personnel safety while working with them. These include (a) test support equipment installation and checkout, (b) system assembly and checkout, (c) launch stand compatibility tests, (d) lander sterilization, (e) flight-instrumentation calibration, and (f) on-stand mating final checks.

Engineering data acquired in accordance with the requirements established above will be processed and analyzed to provide an evaluation of the system performance from launch through planetary encounter.

18. Drop test program. A drop test program will be run. It will include both tower and aircraft drop tests to subject the lander descent system, impact attenuator, and reerection system to anticipated vertical and horizontal combinations in additon to a variety of topographical configurations. The lower drop

test program will be conducted during mid-1966; approximately 20 such tests will be required. The aircraft drop test program will allow evaluation of the drogue stability system and main chute deployment and inflation. If high Mach number deceleration and stabilization tests are required, rocket-boost tests will be conducted.

19. Earth entry test program. Two full-scale Earth entry tests are planned for the Mars lander. These tests will be designed to simulate the entry heating experienced in Mars entry. An entry velocity of approximately 24,000 ft/sec as a reentry angle of 90 degrees will subject the entry vehicle to a maximum deceleration of approximately 150 g. These tests will demonstrate the capabilities of the entire entry system under simulated Martian entry conditions. The aerodynamic, thermodynamic, structural, and descent subsystems will be thoroughly evaluated at high Mach numbers. These flights are planned for the Atlantic Missile Range, existing booster systems such as the Atlas-Agena 30 ks 8000 being suitable for these tests. Three Mars lander vehicles will be manufactured for these tests which include two flight vehicles and a spare. These tests will be run in late 1967 to provide design information early enough in the program to be of use in improving the Mars 1969 entry vehicle design.

20. Qualification test program. Qualification tests will be performed on each different spacecraft configuration. The qualification tests are conducted to demonstrate the ability of the design to meet the performance specifications after being subjected to test levels exceeding the severe operational environments.

The qualification tests will be performed on assemblies, subsystems, and systems of final design.

The qualification tests will encompass various physical inspections and functional tests. Exercise of physical devices and full electrical operation will be performed before, during, and after specified environmental exposures. In this manner, compliance with the specifications during testing may be determined.

The qualification tests will determine various physical and electromagnetic properties, among which are (a) weight and balance (cg), and (b) moment of inertia (including location of principal axes and moments about axes), (c) alignment of attitude control reference axes with principal axes, (d) pressure leakage of any sealed assembly, (e) magnetic properties (including (1) alternating fields and direct fields during all modes of operation, and (2) residual field, including changes), (f) electrical interference (including (1) electrical and radio frequency interaction between subsystems of spacecraft and (2) radiation of signals which are potential interference with launch vehicle or range equipment).

Among the environmental tests included in the qualification testing program are (a) handling shock, (b) transportation shock, (c) acceleration, (d) vibration (mechanical and/or acoustic drive), (e) temperature, (f) humidity, and (h) thermal-vacuum (cold wall, solar simulation where required) and explosive atmosphere.

21. Manufacturing and quality control. The manufacturing and quality control effort for the Voyager Program includes the labor, materials, and subcontractor systems required to produce the hardware listed in table 3. An allowance for manufacturing and quality control attrition of one attrition unit for every three hardware units required is included. Acceptance testing and sterile assembly of the qualification test items and the flight and spare spacecraft is indicated. All subassembly and component level manufacturing and assembly are to be done under clean room conditions.

2.2 Critical Problem Areas

The Voyager design study has revealed a number of design and development areas which represent potential problems in carrying out the Voyager Program. The most important of these are listed and briefly discussed.

1. Sterilization. To comply with the requirement that Mars be kept free from contamination with the probability of 1 in 10,000 requires development of techniques which are beyond the present capabilities of clean-room manufacture and assembly. Not only does the requirement for sterilization impose real problems in the reliability of equipment but also results in major difficulties in demonstration that the desired degree of freedom from contamination has been attained.

2. Surface topography of Mars. One of the purposes of the Mars spacecraft is to obtain more information about the surface topography. Yet in the absence of this information, the lander must be designed so that it has a capability of surviving and communicating with Earth. The proposed solution to this problem is to use a lander with a reerection capability together with a communications relay link. More topographical information would clearly increase the confidence in this or any design.

3. Communications. There are several difficulties in the design of the communication systems. First the signal level received by the lander command link with the DSIF is marginal. Second, uncertainty in the surface terrain may result in multipath transmission from the lander. Third, the possibility of voltage breakdown on the lander antennas limits the transmitted power which can be used.

4. Reliability. The extremely long mission lifetime imposes new challenges to the reliability of all components of the system. The flight time for type II trajectories to Mars is greater than 1 year in 1975. This together with the 6-month mission duration after encounter results in a total mission life of a year and a half.

5. Heat Shield. The design of the heat shield for Mars imposes no particular difficulties. The heat shield design for Venus, however, represents development problems because of the extremely high heat fluxes which will be encountered during direct entry. Adequate facilities for the simulation of these fluxes do not exist and must be developed.

6. Atmospheric variation of Mars and Venus. The uncertainty in the atmospheric model for Mars and Venus and in particular the diurnal and annual variations in the atmosphere require entry vehicle designs which are adaptable to a wide variety of conditions.

7. Instrumentation design. The primary difficulty in scientific instrumentation is to achieve the high reliability and long life required with instruments which can sustain terminal heat sterilization.

8. Radioisotope thermoelectric power supply for the lander. Development of an RTG power supply capable of delivering 110 watts and having a reasonable weight is one of the pacing items for the program, and this activity should be started at once.

9. In-transit thermal control. The same basic orbiter design is planned for both Mars and Venus. The thermal control technique proposed is completely passive and will be achieved by the use of different surface coatings for the Mars and Venus missions. Much more detailed design is required before this approach can be proposed with confidence.

10. Surface environment of Venus. The present lack of information about the surface environment of Venus makes the design of a lander difficult and the design of a direct link capability extremely questionable. The surface temperature also poses severe problems in designing a lander with long mission life.

11. Mapping of Venus. The cloud cover of Venus makes visual mapping impossible and its slow rate of rotation makes mapping of any kind difficult. In order to achieve wide-area mapping, either extremely long lifetime in a polar orbit or mapping from different orbital planes is required.

12. Space environment. Adequate knowledge does not exist at present to design adequate meteroid protection nor to design for the effect of other cosmic particles which may cause sputtering and degradation of materials.

The success of the Voyager program will be dependent in part on our ability to find solutions to these problems. An important part of the Mariner program will be to help provide these answers.

2.3 Major Facilities Requirements

The following major facilities are required for the performance of the Voyager program. These facilities are necessary to subject components, subsystems, and the complete spacecraft to the anticipated environments and to demonstrate the performance capability. The total costs for the facilities are included in the cost estimates.

1. Temperature-humidity chamber.

   Size: 30 by 30 by 30 feet
   Range: 260 to 300° F
   0 to 100 percent R. H.
   For qualification and acceptance testing of full-scale vehicles.

2. Space simulator (includes solar simulator)

Size of Chamber: 30-inch-diameter by 26 feet high
Inside test chamber: 23 feet in diameter by 20 feet high.
Solar simulation beam: 20 feet in diameter.
Nitrogen cooled walls.
Variable solar intensity from 0 to 250 w/ft on target.
Pumping capacity to keep vacuum better than 10^{-5}mm Hg.
Automatic printout equipment for 250 instrument channels.

3. Expansion of vibration facilities

7500 pounds weight, 1g at 16 to 42 cps.
0.011 inch double amplitude 42 to 95 cps.
5 g's at 95 to 2000 cps.

4. Acceleration facilities

Bus-lander	6.25 g axial
	2.0 g axial
Lander only	125 to 340 g axial
	20 to 55 g lateral

5. Hydraulic shaker

7500 pounds weight.
1 g at 1 to 16 cps.

6. Special shock equipment. High-pressure shock facility for component and system development tests.

7. High-pressure facilities. High pressure gas source for attitude control component and system development and qualification testing.

8. Recording equipment. Readout and recording equipment for pyrotechnic, guidance, and retrorocket testing for system development and qualification testing.

9. RFI facilities. Expansion of electromagnetic radiation facilities for development and qualification testing of components and systems. Primary concern is to minimize spurious conductive and radiative energies in the radio frequency bands which may adversely affect operation of the electronic and pyrotechnic systems.

10. Radiation and micrometeorite facility. A 4,000,000-ev electron volt radiation facility for simulation of radiation and micrometeorite environments of interplanetary excursions. For use in the development of instrumentation and control components.

11. High-accuracy acceleration facility. High-accuracy acceleration capability for programmed and control acceleration forces. To be used for development and qualification of components. In addition, for calibration and acceptance testing of acceleration-sensitive control devices.

12. Thermal structural heating facility – quartz lamp banks

80 to 100 Btu/ft-sec
Full-scale lander surface area

13. X-ray room. Extension of the existing nondestructive test X-ray facility is required to accommodate size of structural items.

14. Clean room area/ additional inspection areas. Clean room assembly and inspection area expansion for assembly, calibration, and acceptance testing of sensitive components.

15. Solar simulator/ thermal vacuum facilities. Space chamber with solar simulation (0 to 130 in/ft^2) beam of at least a 2-foot diameter; nitrogen cooled walls. Internal size: 8 foot diameter by 10 feet long. Pumping capacity to keep vacuum better than 10^{-5} mm Hg. For use in development and qualification of subsystems.

16. Sterilization facility (including pilot plant). A pilot plant will be constructed to house the experimentation equipment from which the procedures, techniques, and tests will be developed, as well as to provide personnel training. The final sterilization facility will be comprised of two buildings, one within the other. The interior building will comprise the major assembly facility and will be approximately 15, 000 ft^2. The outer building and environmental protection will be approximately 20, 000 ft^2. A detailed description is included in volume 3.

2.4 Special Test Equipment Requirements

The successful accomplishment of the Voyager Program requires the use of several items of special test equipment. The following are the major identifiable special test equipments:

1. Air-bearing table and astrosimulator. An air bearing-table will be required to physically support the spacecraft during closed-loop dynamic performance tests on the attitude control systems. The table will be installed in an astrosimulator which will simulate the position and brightness of the sun, Canopus, Mars, and the Earth during the tests. The capacity of the air-bearing table will be sufficient to take the weight of the spacecraft. The size of the astrosimulator will be large enough to accommodate the air-bearing table with the

spacecraft mounted and also permit angular displacement with respect to Canopus, the sun, and Mars.

2. Convective arc. A convective arc facility will be required to evaluate the effect of convective heat transfer to the candidate heat shield materials under a variety of heat fluxes, enthalpies, shear forces, and atmospheres. Convective heat fluxes of several hundred Btu/ft^2-sec must be simulated for the Mars entry. Much higher fluxes must be simulated for the Venus entry. The arcs must be capable of enthalpies up to at least 10,000 Btu/lb. The arcs must also have a wide shear force capability and working gas capabilities of carbon dioxide, nitrogen, and argon in any combination.

3. Radiant arc. For the Venus entry simulation, the capability of attaining heat fluxes up to at least 10,000 Btu/ft^2-sec is required. The requirements for the Mars entry simulation are less severe. Various atmospheres of carbon monoxide, nitrogen, and argon gas mixtures must be available. These arcs will probably be arc-imaging furnaces for the lower heat fluxes and plasma arcs for the higher fluxes. They will be used to establish the basic heat-transfer mechanisms of candidate materials when exposed to radiant energy. The data will be employed to formulate analytical expressions that describe material response to radiant heating and permit heat shield material evaluation.

4. Convective radiant arc. A convective radiant arc will be required to generate combined convective and radiant heat pulses over the ranges of heat fluxes, enthalpies, shear forces, and atmospheres that may be encountered during Mars and Venus entry condition. They probably will be composed of low- and high-intensity, arc-imaging furnaces superimposed over convective heating plasma arcs of varying heat flux, enthalpy, shear, and atmospheric capabilities. The data will be used to generate design information for thermal analysis and also to verify the basic theories of material response to combined convective and radiant heat pulses that are derived from material testing on separate convective and radiant heating facilities.

5. Space simulator devices and associated equipment. A variety of equipment is required to evaluate the effect of the space environment on candidate material for the Voyager vehicles. These include solar radiation, particle radiation, and meteorological impact simulators, used in conjunction with temperature and high-vacuum chambers. In addition, there is the requirement for simulating the planetary environments of Mars and Venus for the evaluation of material properties during exposure to dust storms, solar irradiation, low and high pressure, and temperature extremes.

6. Miscellaneous equipment in support of aerodynamic, thermodynamic, and structural test programs. A variety of equipment and devices are required in support of the multitude of testing necessary for the development of

the vehicle structures and components. Included are (a) special recording, measuring, and instrumentation devices; (b) jigs and fixtures for large vehicle pressure testing and testing of spherical caps and cylinders; (c) equipment to simulate heating conditions; (d) temperature probes, (e) static pressure probes, (f) hot wire anemometry measuring devices; and (g) improvements in dynamic stability instrumentation.

7. Equipment for inspection and test in support of quality control and testing program. A wide variety of electrical, mechanical, and optical equipments and devices are required for performing the multitude of different types of inspections, measurements, and tests associated with the quality control and testing program. The equipments will be utilized during the inspection and testing of components, subassemblies, subsystems, and completed vehicles, as well as functional and acceptance tests.

8. Modification of hyperaltitude special test equipment. Modification of hyperaltitude capability includes the addition of solar simulation devices and modification of existing shroud and tank for altitudes to 10^{-8} mm Hg.

9. Special combined vibration, acceleration, temperature, and altitude test equipment. A special combined vibration, acceleration, temperature, and altitude equipment will be used for development and qualification of guidance and instrumentation systems.

10. Special test support equipment. Included herein is the expansion of vibration recording equipment from 56 to 200 channels and noise analysis equipment to 1-cps bandwidth. The equipment will be used in support of large-scale vibration testing and analysis of full-scale vehicle and systems.

11. Special motion X-ray equipment. The motion X-ray recording equipment will be used as an aid in the development of potted components and systems under vibration.

12. Simulated communications ground station. The TM ground station simulator consists of an S-band automatic phase control receiver, an S-band transmitter, a telemetry demodulator, a synchronizing subsystem, a command subsystem detection and demodulation equipment, and associated power supplies. It is required to provide a radio link checkout capability for the spacecraft at various times during the development and during launch sequences.

2.5 Summary of Major Subcontracted Items

Development of several of the Voyager subsystems will be subcontracted. In certain cases, only specialized components will be subcontracted. A list of the anticipated major subcontracts consists as follows:

1. Communications. Subcontracts will be issued for the design, development, test, manufacturing, and assembly of all of the following major equipment comprising the communications systems for both the orbiters and landers (a) radio subsystem direct link, (b) telemetry subsystem, (c) command subsystem, and (d) relay link subsystem.

2. Power sources. Subcontracts will be issued for the design, development test, manufacturing, and assembly of the equipment necessary to furnish, convert, and distribute the power required during the spacecraft operation. Power for operation of the orbiters will be furnished by solar cells supplemented by batteries and associated equipment. For the Mars lander, power will be supplied by a radioisotope thermoelectric generator supplemented by batteries. Power to the Venus lander will be provided by batteries.

3. Guidance subsystems and stabilization control subsystem. Subcontracts will be issued for the design, development, test, manufacturing, and assembly of the equipments comprising the guidance and stabilization control subsystems. For the orbiters, the following is included:

   a. Guidance.

      1) Accelerometer package and associated equipment

      2) General-purpose digital computer

      3) Auxiliary star tracker

      4) Planet tracker

      5) Planet horizon scanner.

   b. Stabilization and control subsystem.

      1) Gyro package and associated electronics

      2) Cold gas reaction system

      3) Coarse and limit cycle sun sensors

      4) Canopus tracker.

For the lander, a digital computer and sun sensor are included.

4. Mapping. Subcontracts will be issued providing for the design, development, test, manufacturing, and assembly of the following systems capable of mapping the planets to a high degree of resolution:

a. Optical mapping system--Mars orbiter

b. Radar mapping system--Venus orbiter.

5. Propulsion. A subcontract will be negotiated for the design, development, test, and production of the engine, fuel, and associated hardware comprising the propulsion systems for operation of both the orbiters and landers.

6. Descent system. A subcontract will be issued for the design, development, test, manufacturing, and assembly of a descent system for the landers. The descent system is comprised of a parachute-actuation system armed by rising entry acceleration and triggered at a preset g level, ejecting a drogue parachute at Mach 2.5 by the use of a gas generator and mortar; and a main parachute, which is deployed by either a barometric switch or a radar altimeter signal, at Mach 0.8 or less at an altitude greater than 15,000 feet.

7. Aerodynamic wind tunnel tests. Subcontracts for conduction of the wind tunnel test program for the landers will be issued to facilities that are capable of providing large variations in Mach number, Reynolds number, and atmospheric composition. Stability test studies are required to provide confirmation of analysis of aerodynamic loads, heating, and ablation characteristics of the proposed entry body configuration.

8. Reliability testing. Reliability demonstration testing of certain major subassemblies will be subcontracted to the vendor site to provide special test facilities for such tests as are unavailable at the prime contractor's facility. The subcontractor will provide both test samples and test operations to be closely controlled by prime contractor personnel.

2.6 Long Lead and Delivery Items

The studies on the Voyager program have highlighted several areas of development and procurements, the lead times on which require prompt initiating action coupled with close monitoring to ensure successful completion for the Mars 1969 window. Three areas of development exist, however, wherein the lead time required is considered sufficiently critical to justify immediate initiation of a development program. These areas are (1) sterilization, (2) the RTG fuel element, and (3) the scientific instrumentation.

1. Sterilization. Each flyable lander will be subjected to a sterilization procedure to ensure compliance with NASA's "germ-free" requirements. To determine, test, and demonstrate the sterilization approach to be used, a pilot plant operation is being proposed as the first step. From the pilot plant operation, the procedures, techniques, and data to obtain the necessary sterilization confidence levels would be determined. Approximately 32 months are required from the initiation of the pilot plant facility to the availability of a formal sterilization facility. During this period, the planning, data gathering, testing, staffing, training, and purchasing requirements for the entire program will be performed.

2. RTG fuel element. It is presently anticipated that the fuel element for the radio-isotope thermoelectric generator will be plutonium 238 (Pu-238). The development and delivery lead time for Pu-238 is approximately 2 years. The alternate fuel element for the RTG power source being considered is curium 244 (Cm-244). Cm-244 has a shorter development and delivery lead time, but because of its higher radiation background, is considered a less attractive choice for the power source fuel element.

3. Scientific instrumentation. From the nature of the mission as well as the environmental conditions to which they will be subjected, it is apparent that the scientific instrumentation to be assembled into the orbiters and landers will be unique. It is anticipated that the minimum lead time for the development and delivery of the first science packages for the Mars 1969 launch may be approximately 3 years. Included in this period is the determination of mission concept, the selection of instrument type, the issuance of design contracts, development, testing, and manufacturing of the hardware units.

2.7 Government-Furnished Equipment, Items, and Services

The following lists the items that will be Government-furnished. The cost for these items and services has not been included in the Avco estimates.

1. Boosters. Two Atlas-Agena-30Ks 8000 boosters will be required for the conduct of the Earth-entry tests scheduled for the last quarter of 1967. The Earth-entry tests will be conducted only for the Mars lander vehicle. Two Saturn boosters will be required for each launch opportunity. The total number required is 16.

2. Scientific instrumentation. Instruments to be included in the orbiter and lander(s) for each Voyager flight, in accordance with the mission concept, will be required for 24 orbiters (12 each for the Mars and Venus launches) and 42 landers (18 for Mars and 24 for Venus launches).

The science packages for the Mars 1969 launch opportunity are required by April 1967 in order to be integrated with and assembled into the first units undergoing qualification tests.

3. RTG fuel element. The power source for the landers to be used on the Mars launches will be a radioisotope thermoelectric generator. The fuel element to be selected will be either plutonium 238 (Pu-238) or curium 244 (Cm-244). Of the two, the former is more desirable because of its lower radiation background, but less attractive because of its longer lead time. A total of 18 of the fuel elements will be required for the program. Delivery of the three elements for the Mars 1969 launch window is required by April 1967.

4. Aircraft for drop tests. Airdrop tests will be conducted to demonstrate the design of the Mars and Venus lander descent and impact system. These tests will be in addition to the tower drop tests. It is expected that NASA will provide the instrumented aircraft and support services to perform these airdrop tests.

5. Range support. It is expected that NASA will provide for each launching, the launch facility, downrange tracking, and the DSIF support including the services and equipment associated with each.

6. Data analysis. It is expected that NASA will furnish the facility and equipment to receive, decode, and store the data transmitted back from each launch firing. NASA is further expected to provide the service of interpreting and analyzing the data as reduced by Avco.

A summary of hardware requirements for the Mars and Venus launches is included in table 3.

TABLE 3

SUMMARY OF HARDWARE -- MARS AND VENUS LAUNCHES
Mars Launches

Mars Window	Development Models		Mockups		Structural Test Units		Drop Test Units		Earth-Entry Test Units		Qualification Test Units		Flight Units		Subsystem Spares Flight Units (sets)	
	O	L	O	L	O	L	O	L	O	L	O	L	O	L	O	L
1969	2	2	5	5	3	3	0	6	0	3	2	2	3	3	1	1
1971	1	2	-	-	-	-	-	-	-	-	1	2	3	6	1	2
1973	-	-	-	-	-	-	-	-	-	-	-	-	3	3	1	1
1975	-	-	-	-	-	-	-	-	-	-	-	-	3 buses	6	1 bus	2
Totals	3	4	5	5	3	3	0	6	0	3	4	4	9 orbiters 3 buses	18	3 orbiters 1 bus	6
GSE (Mech)											2 Qualified Units		8 operational units			
GSE (Elec)											2 Qualified Units		6 operational units			

O = orbiter
L = Lander

TABLE 3 (Concl'd)

Venus Launches

Venus Window	Development Models			Mockups			Structural Test Units			Drop Test Units			Earth-Entry Test Units			Qualification Test Units			Flight Units			Subsystem Spares Flight Units (sets)		
	O	C	L	O	C	L	O	C	L	O	C	L	O	C	L	O	C	L	O	C	L	O	C	L
1970	1	3		5	5		3	6		0	6		0		0	2	6		3	9		1	3	
1972	1	3								0	6								3	9		1	3	
1973	1		1			5			2	0		2				2		2	3		3	1		1
1975																			3		3	1		1
totals	3	6	1	5	5	5	3	6	2	0	12	2	0	0	0	4	6	2	12	18	6	4	6	2

O = Orbiter

L = Lander

C = Capsule

3. SCHEDULES

3.1 Mission Schedule

Mission schedules for all Mars and Venus launch opportunities between 1969 and 1975 are summarized in table 4. This table shows milestones for flight unit delivery to Cape Canaveral, launch dates, enroute intervals, and planetary operation intervals and is designed to reveal any launch opportunity overlaps which might cause interferences in launch pad availability or DSIF traffic saturation. As indicated in table 4, there are no severe overlaps of launch pad requirements between any two launch opportunities. There is a slight overlap of a few months between the Venus 1975 and the Mars 1975 opportunity, but this is not great enough to cause any pad availability problems. The launch pad facilities which will be available for the Saturn S-1B and 1969 will accommodate a maximum of two simultaneous launchings, which is just adequate for the recommended program of two launchings for every opportunity.

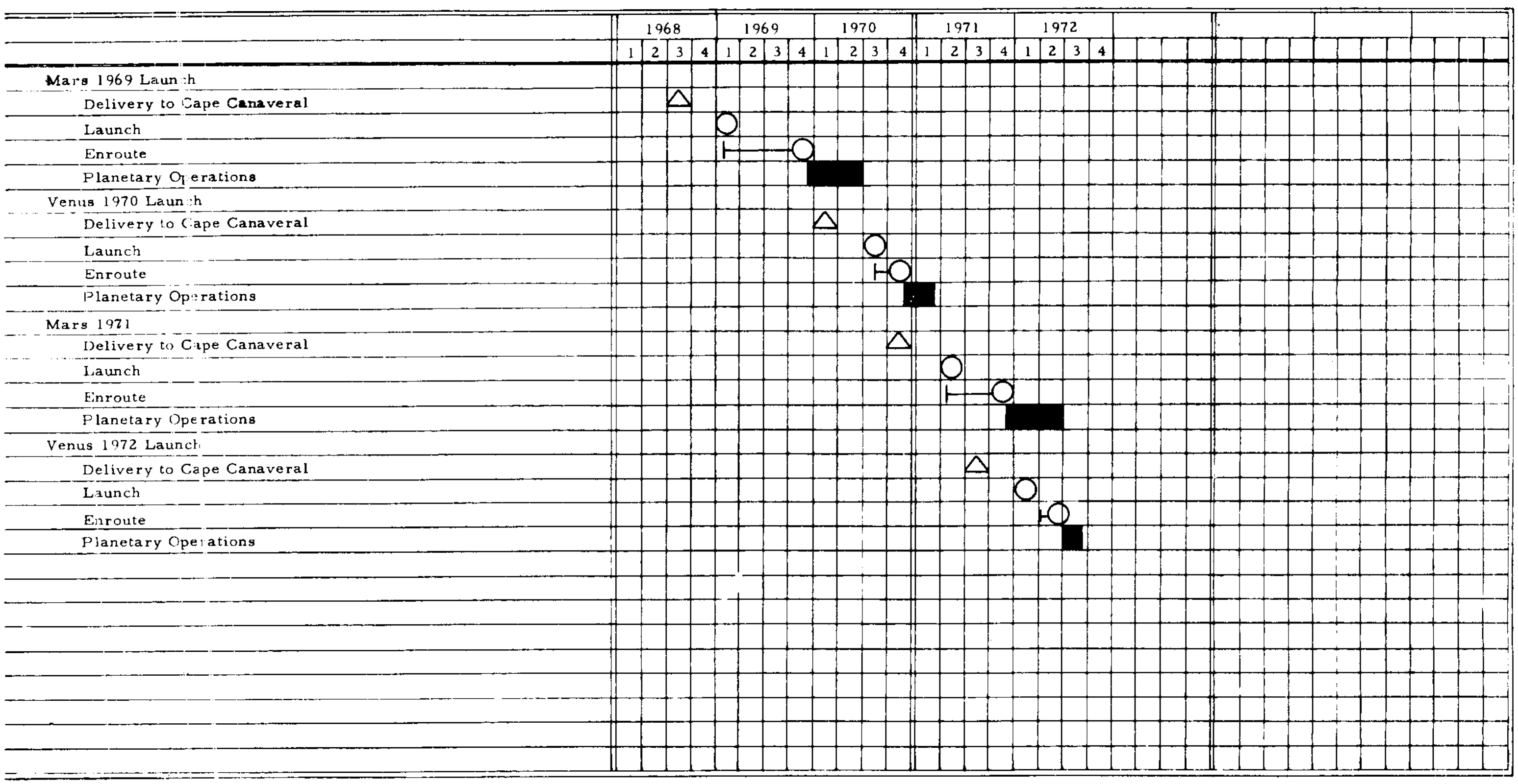
The schedule also indicates that there are no serious mission overlaps which might create a DSIF traffic saturation problem. There is overlap between planetary operations and enroute intervals for several adjacent opportunities. The Venus 1972, an enroute interval, overlaps the Mars 1971 planetary operations interval and the Venus 1973 enroute and planetary operations intervals. The DSIF facility has a maximum tracking capability for two simultaneous flights. In the case of the overlaps, there will be a maximum of four simultaneous flights if all launchings are successful, but the look angles to Mars are significantly different than the look angles to Venus such that the DSIF will be tracking a maximum of two simultaneous flights which is within its capability. There may be short-term exceptions to this during and immediately after a launch, at which time it may be necessary to delay tracking of an in-transit flight for a short, but acceptable, interval.

3.2 Development Schedule

The reference Development Plan schedule (Plan I) is shown in detail in table 5 and summarized in table 6. The reference Development Plan (Plan I) schedules initiation of the hardware contract in the last quarter of 1964 to meet the first launch opportunity of Mars 1969. An alternate Development Plan (Plan II), shown in table 7 slips the initiation of the hardware contract to the middle of 1965. The reference plan was set up to minimize overlap of the normally sequential development functions to provide a reasonable noncrash program. Plan II compresses the time available for design, manufacturing, and testing and causes overlap between development testing and qualification testing. This will require more extensive employment of overtime labor and increase the possibility of crash modification recycling if the development tests indicate a requirement for design changes.

TABLE 4

MISSION SCHEDULES



LEGEND:
Date on which spacecraft to be delivered
Scheduled launch window
Period enroute to planet
Period of planetary operations

TABLE 4 (Concl'd)

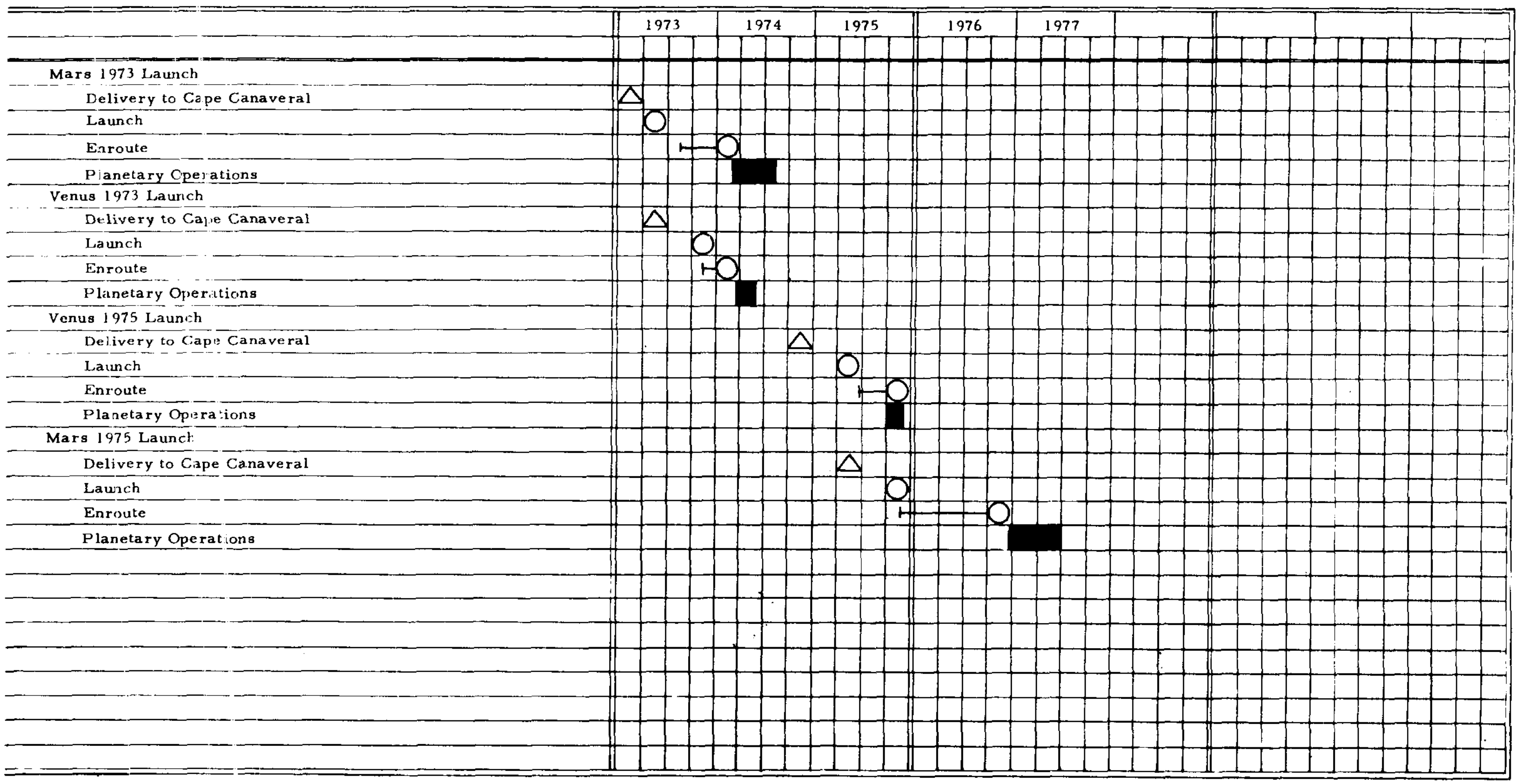


TABLE 5

VOYAGER DEVELOPMENT ~~PLAN~~ — MARS 1969 LAUNCH

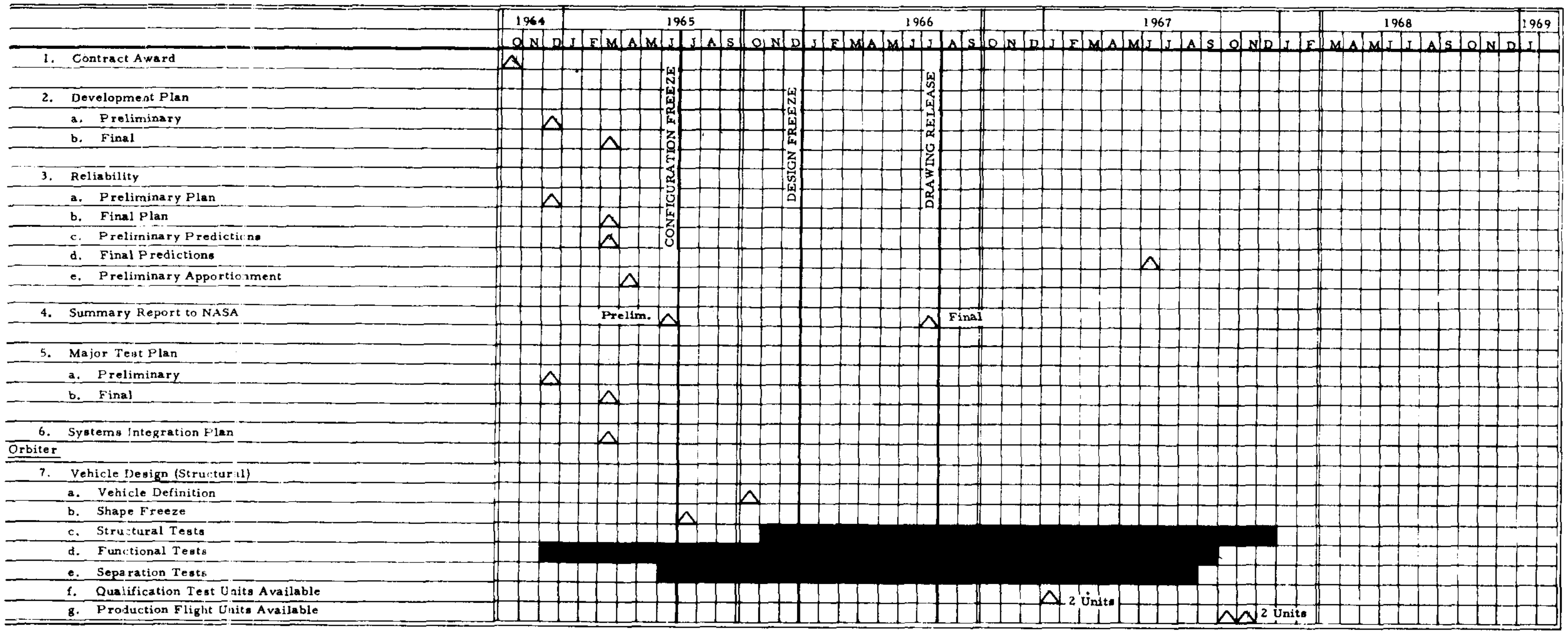


TABLE 5 (Cont'd)

3.5 Voyager Development Plan -- Mars 1969 Launch (Cont'd)

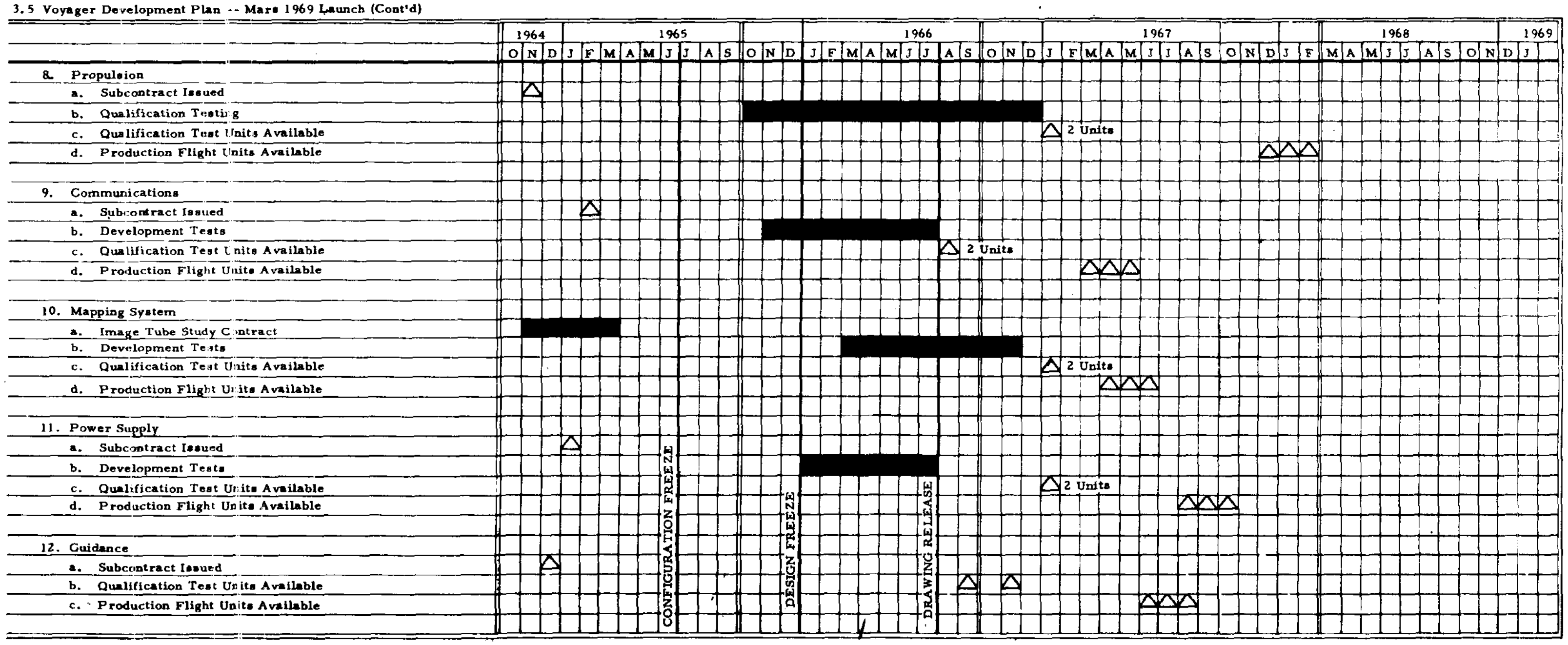


TABLE 5 (Cont'd)

3.5 Voyager Development Plan -- Mars 1969 Launch (Cont'd)

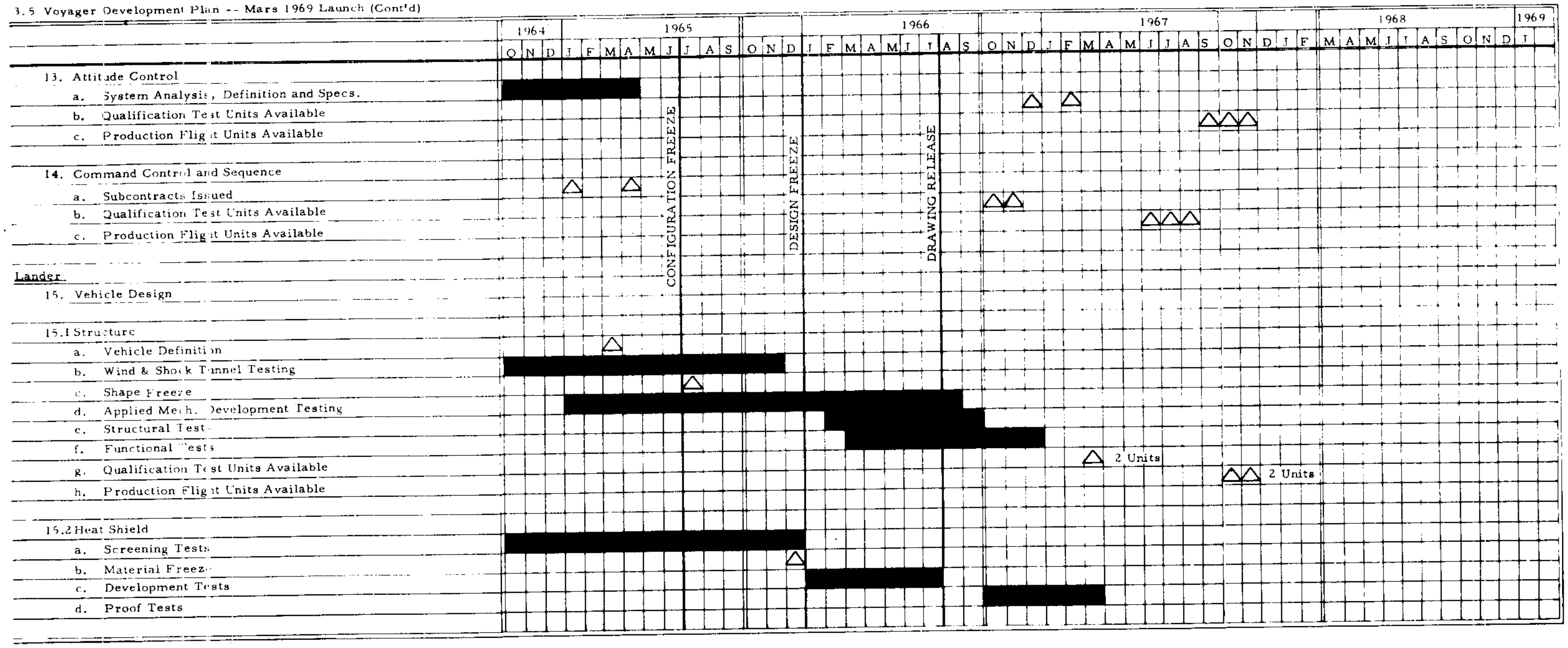


TABLE 5 (Cont'd)

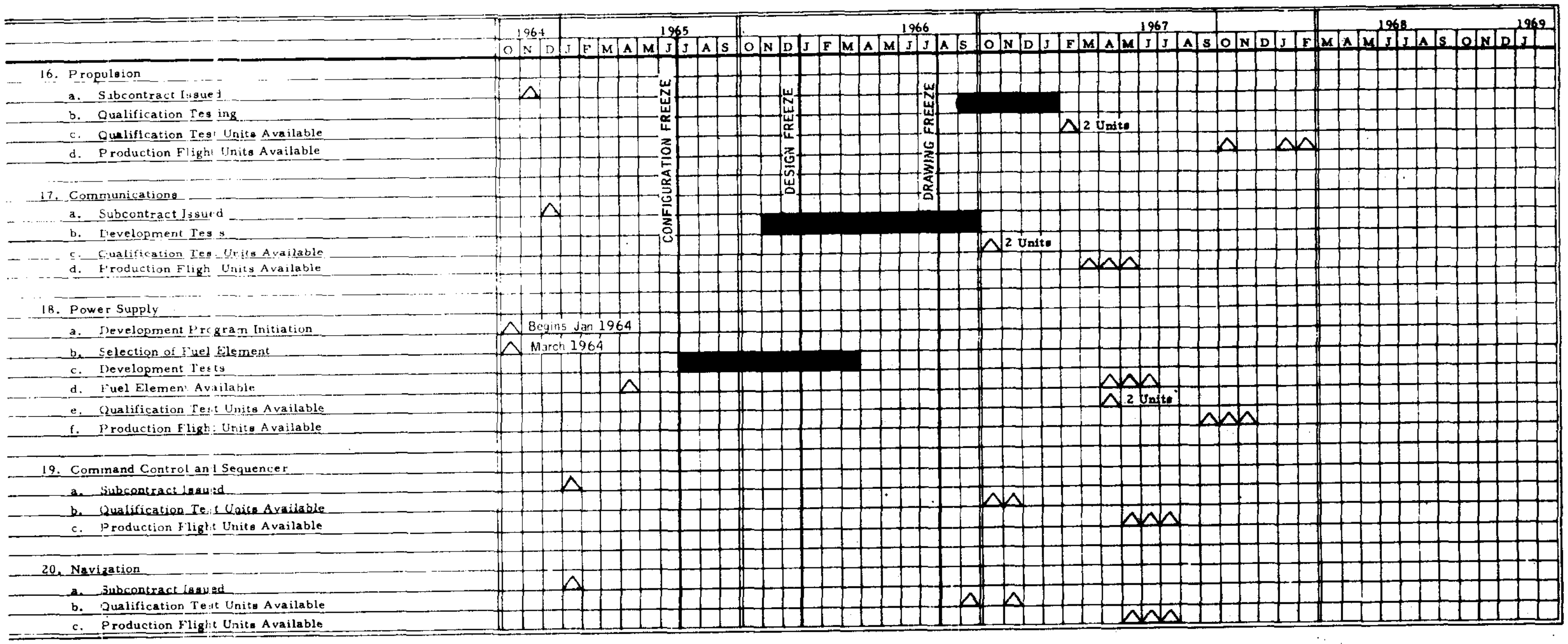


TABLE 5 (Cont'd)

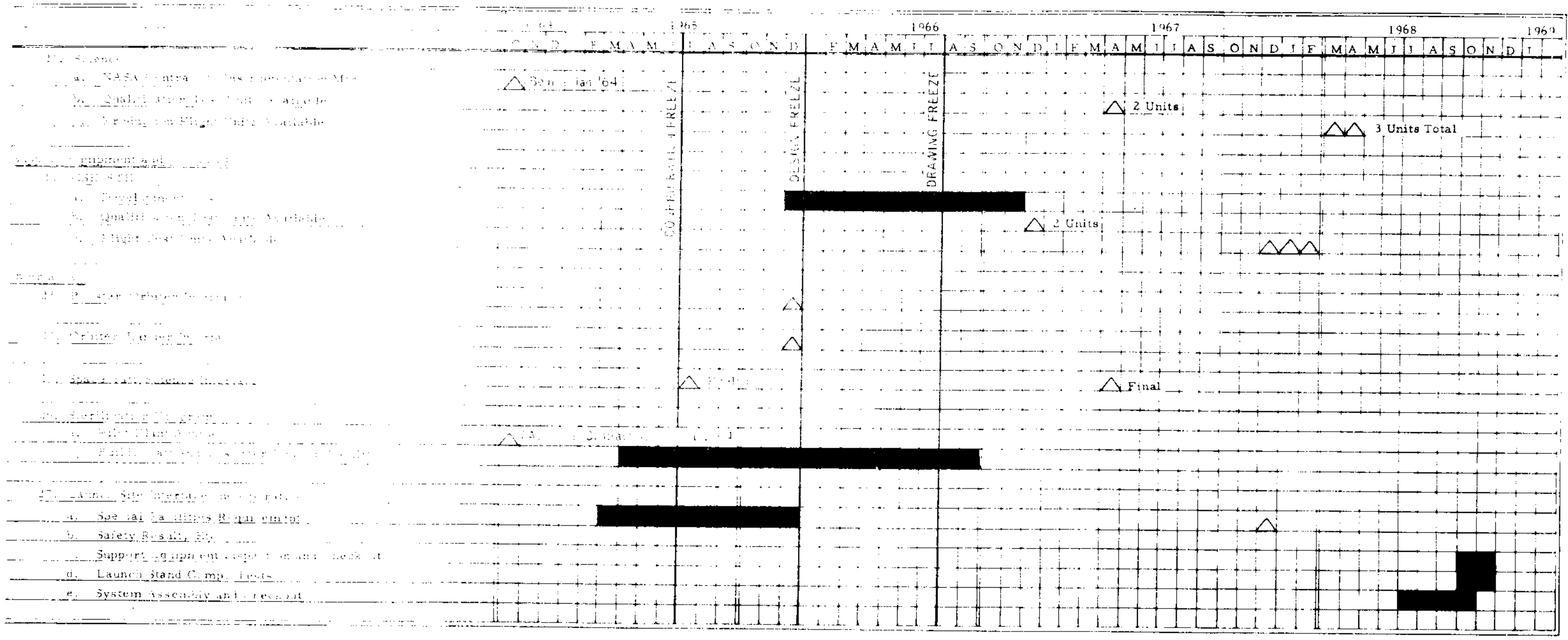


TABLE 5 (Concl'd)

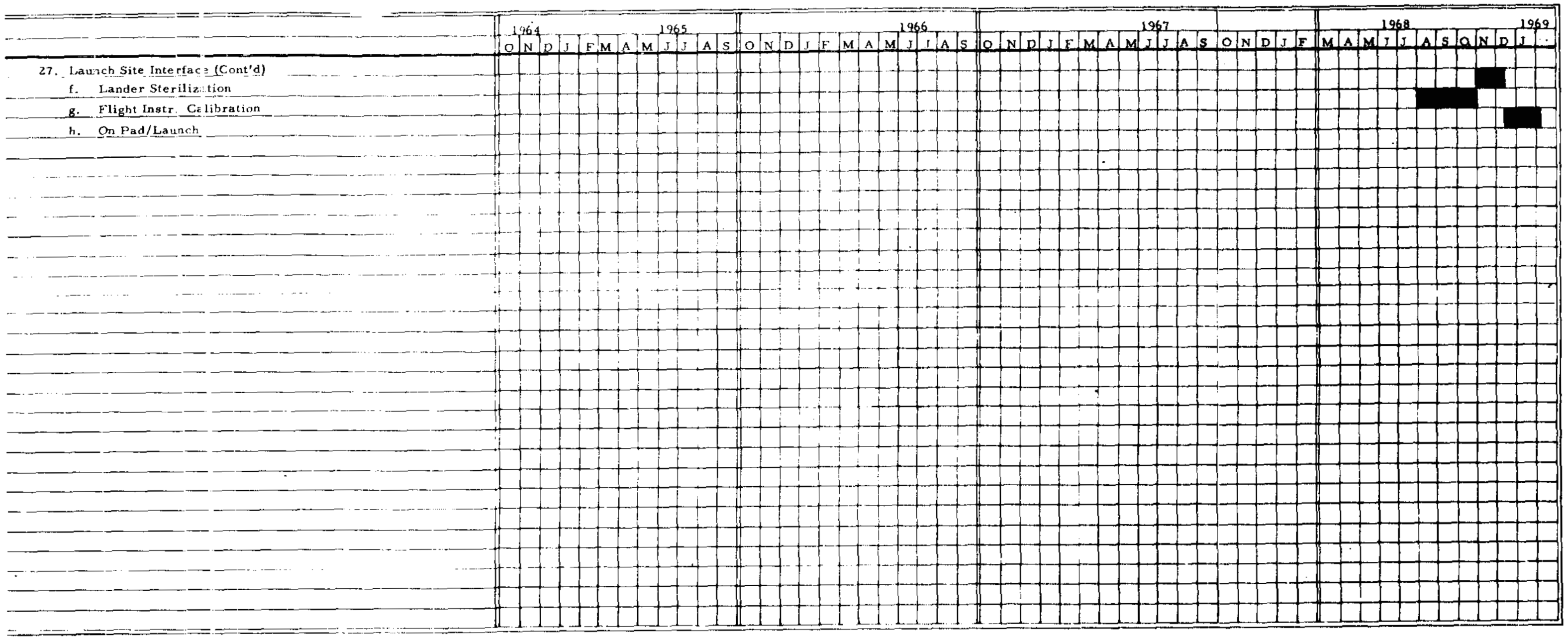


TABLE 6

VOYAGER SUMMARY DEVELOPMENT PLAN--MARS 1969 LAUNCH

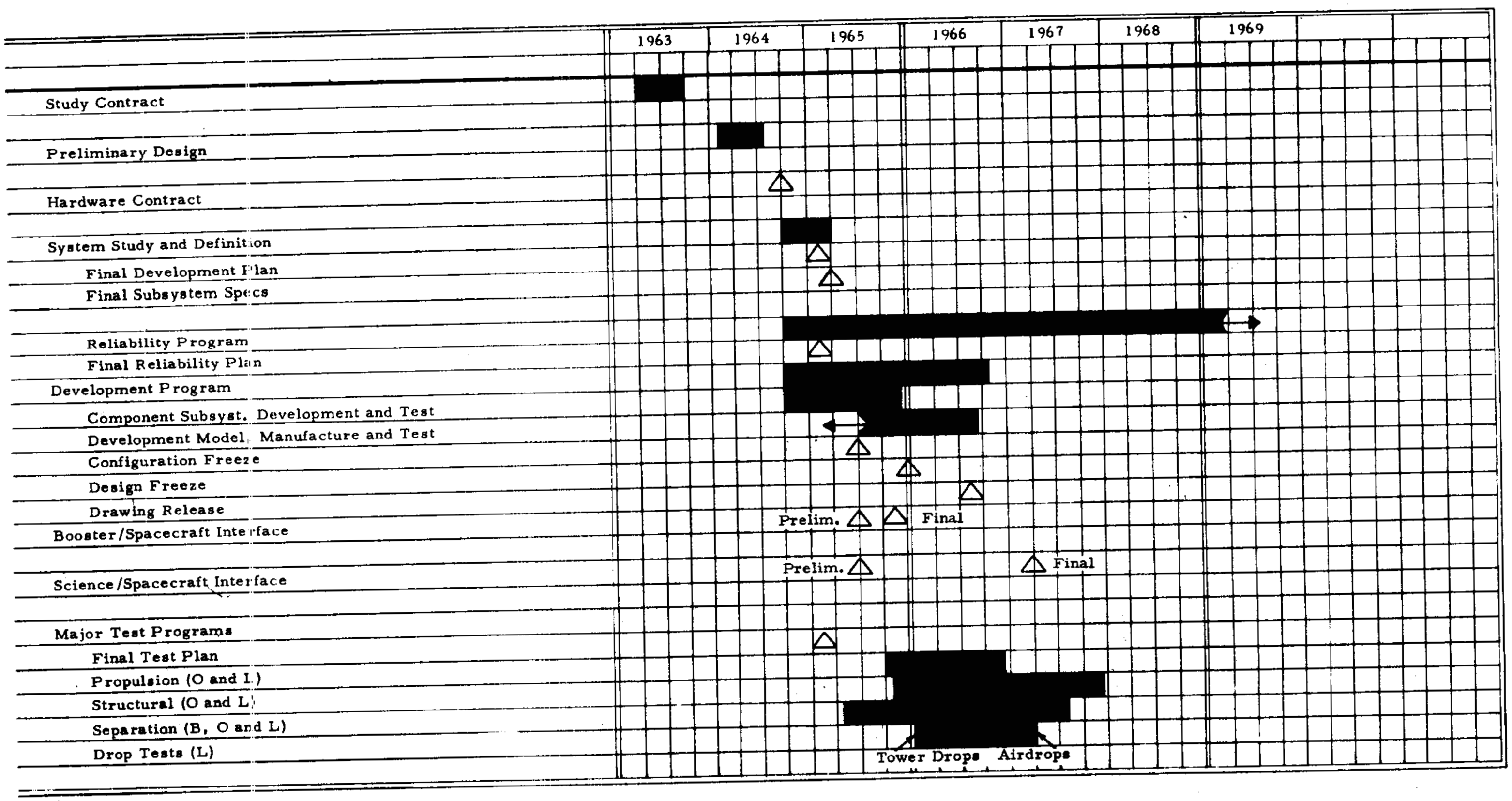


TABLE 6 (Concl'd)

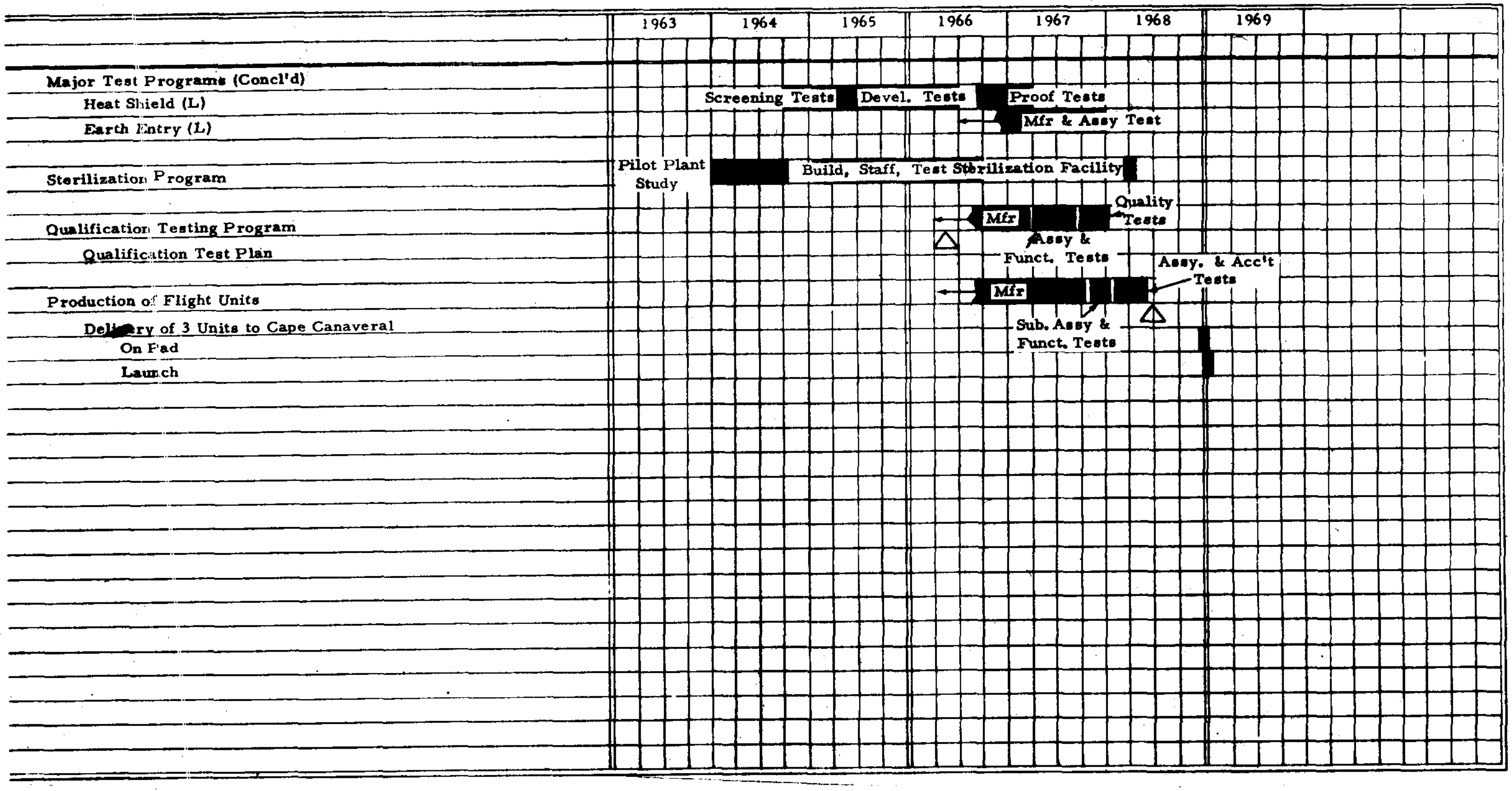


TABLE 7

VOYAGER SUMMARY DEVELOPMENT PLAN II--MARS 1969 LAUNCH

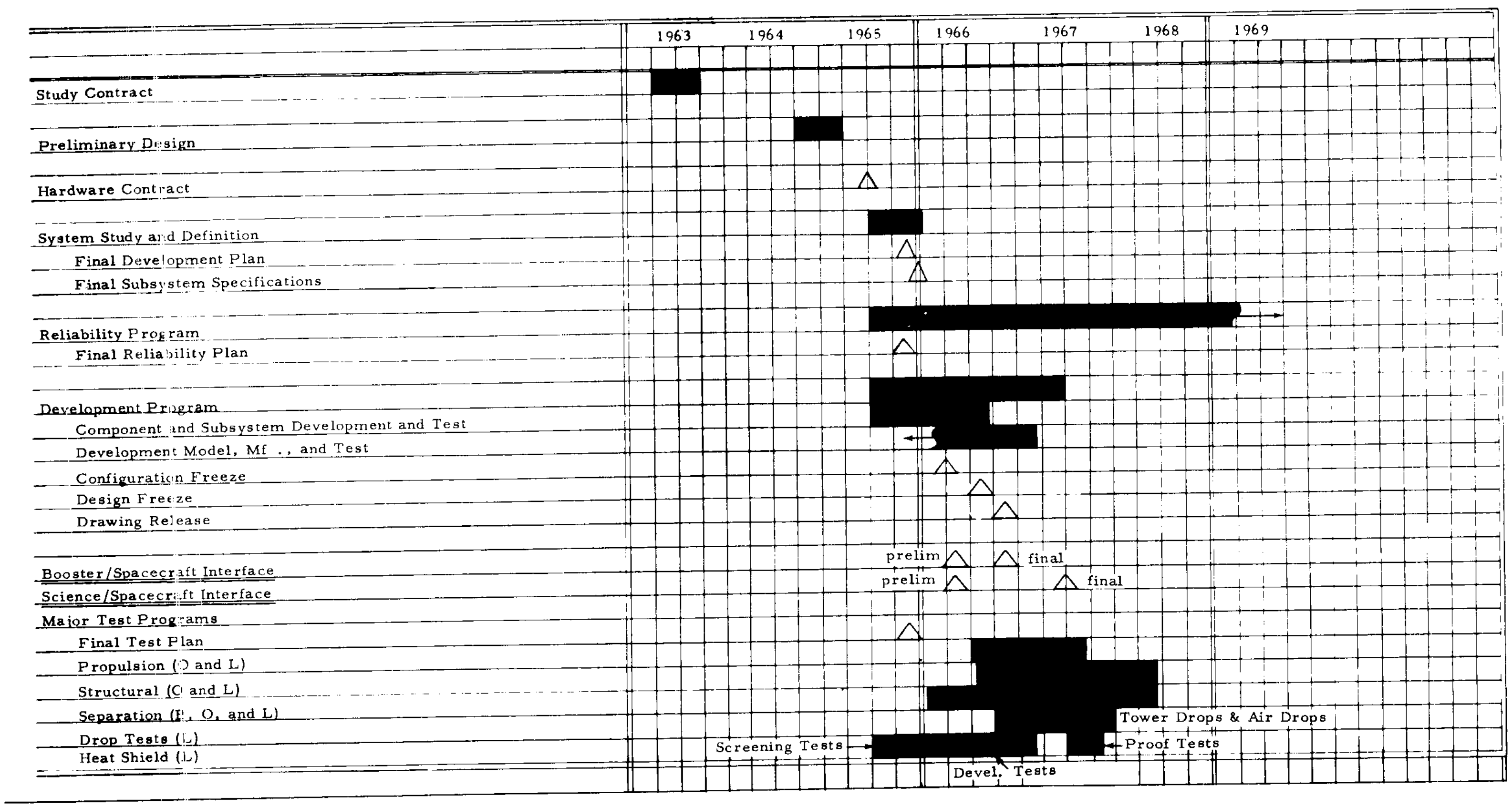
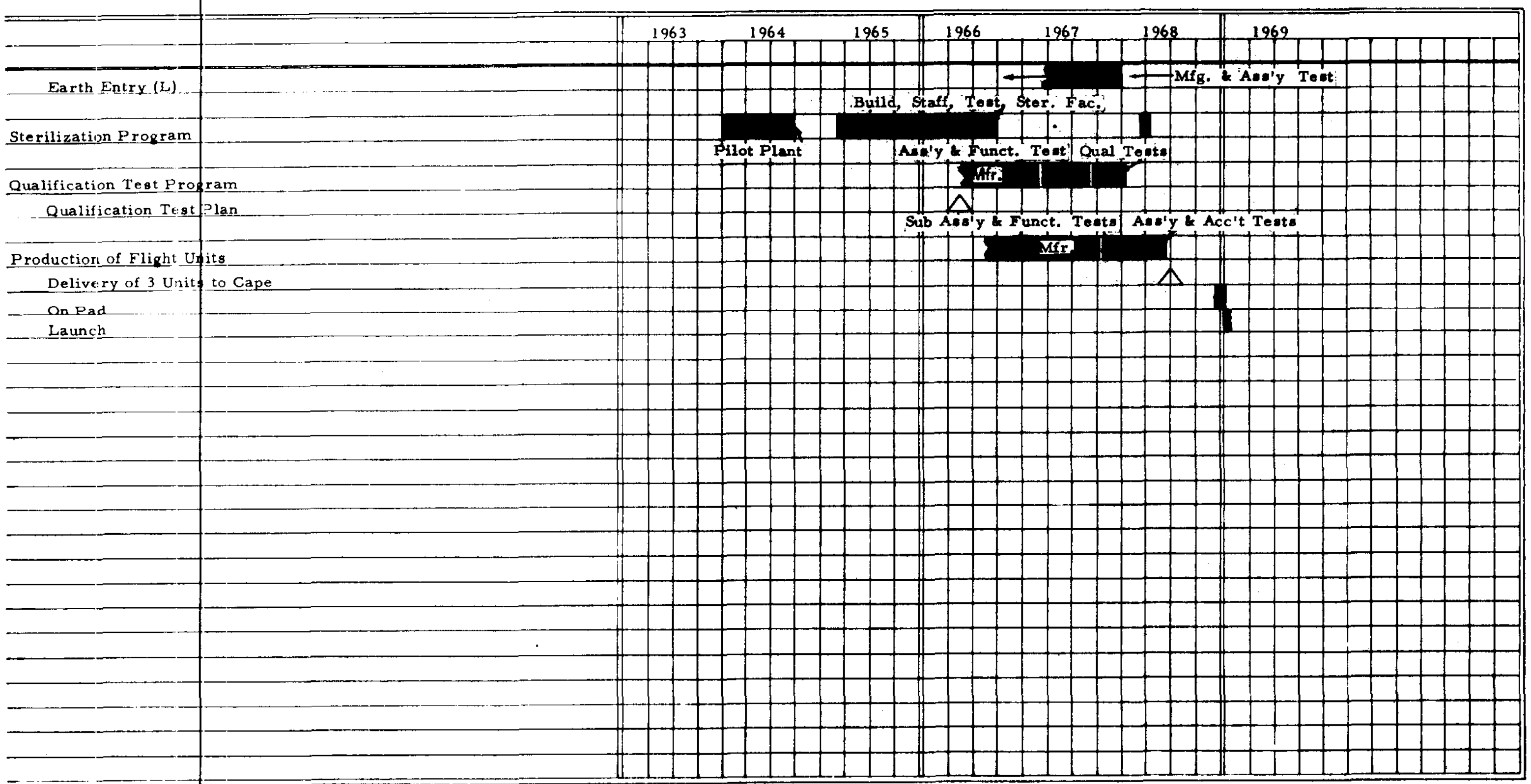


TABLE 7 (Concl'd)



The reference plan was designed to provide optimum phasing of the pacing items such as design, development testing, qualification testing, and manufacturing. These need to be sequential or nearly sequential functions because the progress of one is dependent on the results of the previous item. As can be seen in the summary schedule and the detailed schedule, overlap of these items is minimized to permit a flow forward of design and test results and a recycling backwards of necessary changes on a noncrash basis.

System definitions and final subsystem specifications are scheduled to be completed 6 months after contract go-ahead, and all subcontracts will be issued within 6 months of program initiation. This is based on the expectation that a great part of the system analysis and subsystem evaluation will have been accomplished in a 6-month preliminary design study which precedes the hardware contract. Preliminary and final drafts of the development plan, reliability program plan, major test program plan, and systems integration plan will be completed at contract go-ahead plus 3 months and plus 6 months, respectively.

The design of the vehicle and subsystems will be essentially controlled timewise by three key design milestones: a configuration freeze at contract plus 9 months, a design freeze at contract plus 15 months, and a drawing release at contract plus 22 months (third quarter 1966). The configuration freeze is the date on which the last of the external interfaces is frozen. External interfaces may be defined as the vehicle and subsystem external geometry, weight, electrical connections, installation data, and so forth. The design freeze is the date on which the last of the layout drawings of both subsystems and vehicle are completed. The drawing release is the date on which the last of approved product drawings and specifications is released for manufacturing. Controlled engineering changes are required after drawing release.

Planning and procurement for the development tests begin early in 1965 for most of the tests, as shown in the major test program schedule (table 8). Planning and procurement for some of the later tests will start in the last quarter of 1965. Tests which do not require complete vehicle and subsystem assemblies, such as the six tower-drop lander mock ups, the six airdrop lander mockups, and subsystem functional test models, are scheduled to take place through 1966, after the design freeze. Tests which involve the complete vehicle, such as orbiter and lander structural tests, thermal vacuum tests, separation tests, and the Earth entry tests, will be accomplished through 1967, necessarily about 6 to 9 months after the drawing release. These tests utilize complete vehicles, and sufficient time must be allowed after the completion of detailed design (drawing release) to permit fabrication and assembly of the complete test prototype. Some of these later development tests occur just prior to the qualification test program and are therefore pacing items in the overall development plan. A few development tests, such as arc and heat-transfer tests, wind and shock tunnel tests, and material screening tests, provide basic design data and do not utilize full-scale mockups or test prototypes and will be initiated early in the program, a few months after contract go-ahead. After the drawing release completion in

TABLE 8

MAJOR TEST PROGRAMS--MARS ORBITER AND MARS LANDER

ORBITER

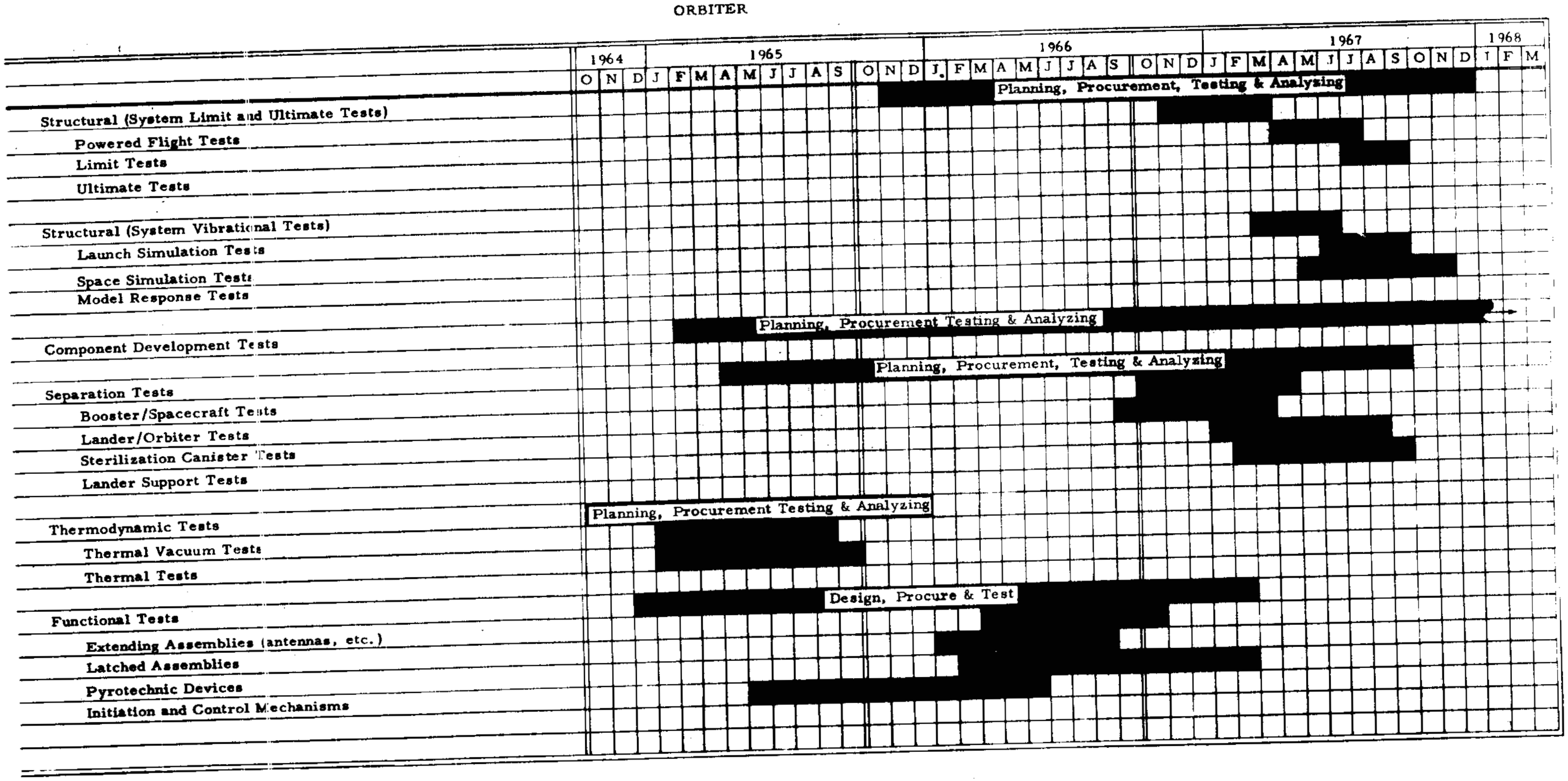


TABLE 8 (Cont'd)

ORBITER (Concl'd)

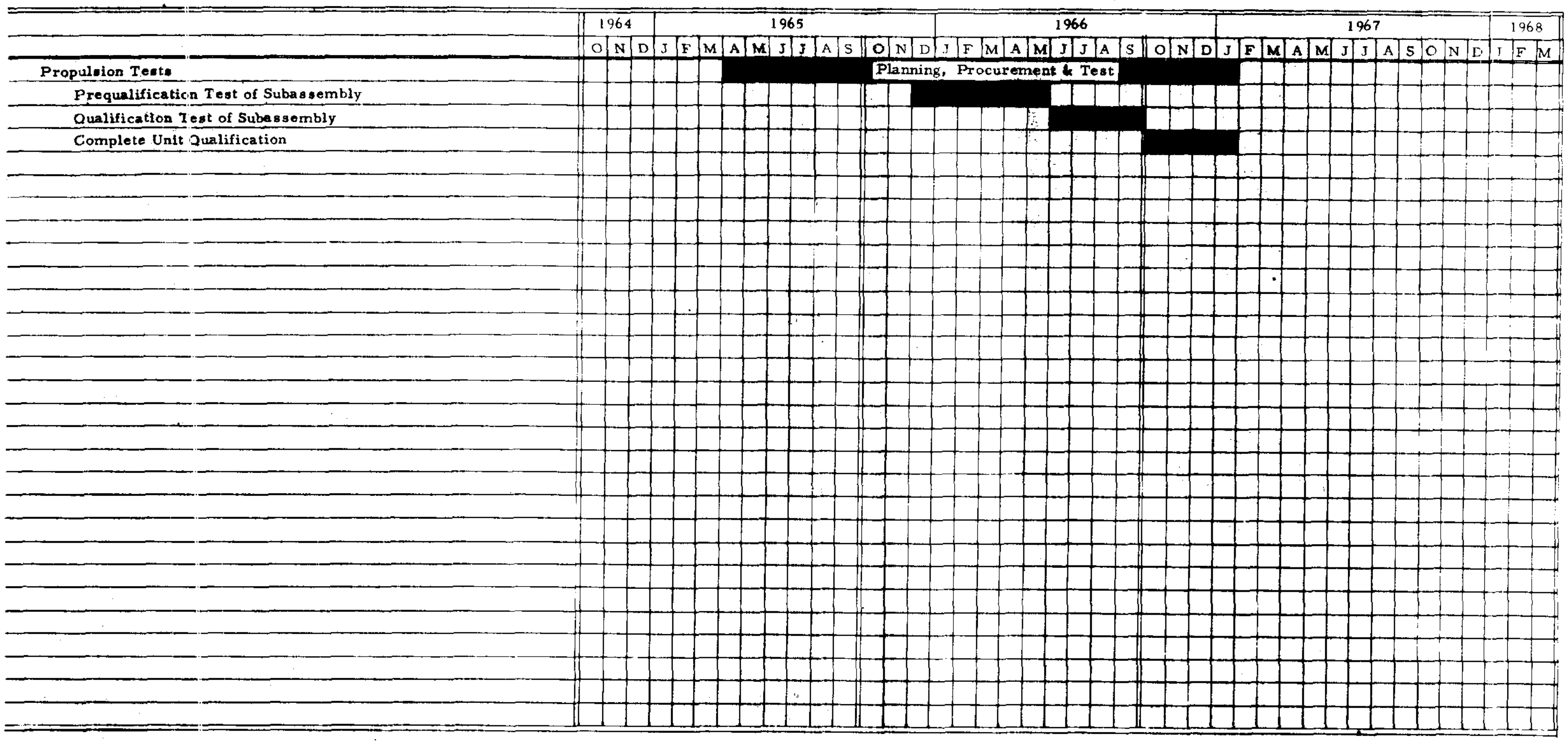


TABLE 8 (Cont'd)

LANDER

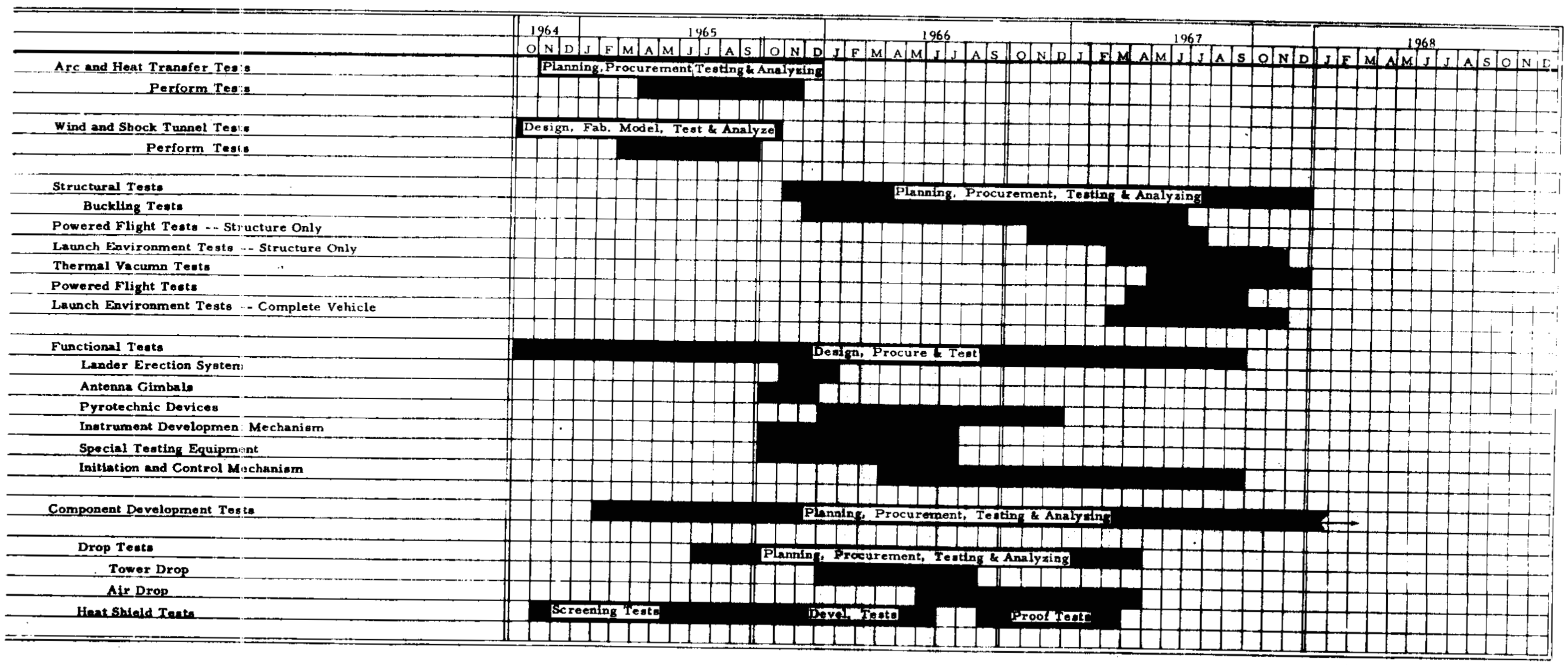
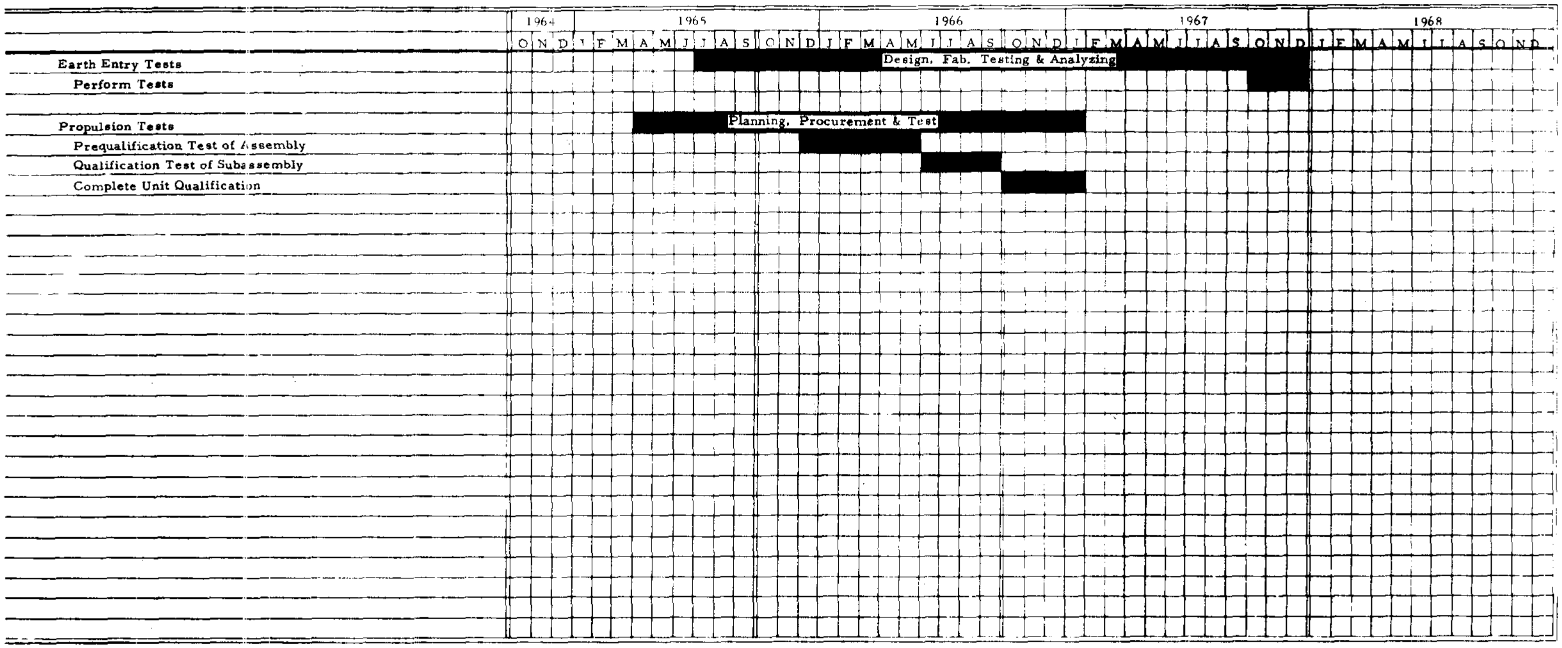


TABLE 8 (Concl'd)

LANDER (Concl'd)



the third quarter of 1966, a maximum of 9 months is provided for fabrication and final delivery of subsystem and vehicle qualification test units in the first quarter of 1967. Some subsystem units will be delivered in the last quarter of 1966. Assembly and functional tests of the qualification prototypes begin near the end of the first quarter of 1967, and 6 months are allocated for this function. Qualification tests begin in the third quarter of 1967 and are completed 5 months later in the first quarter of 1968. Two qualification prototypes of both the orbiter and lander are provided to help expedite the testing schedule.

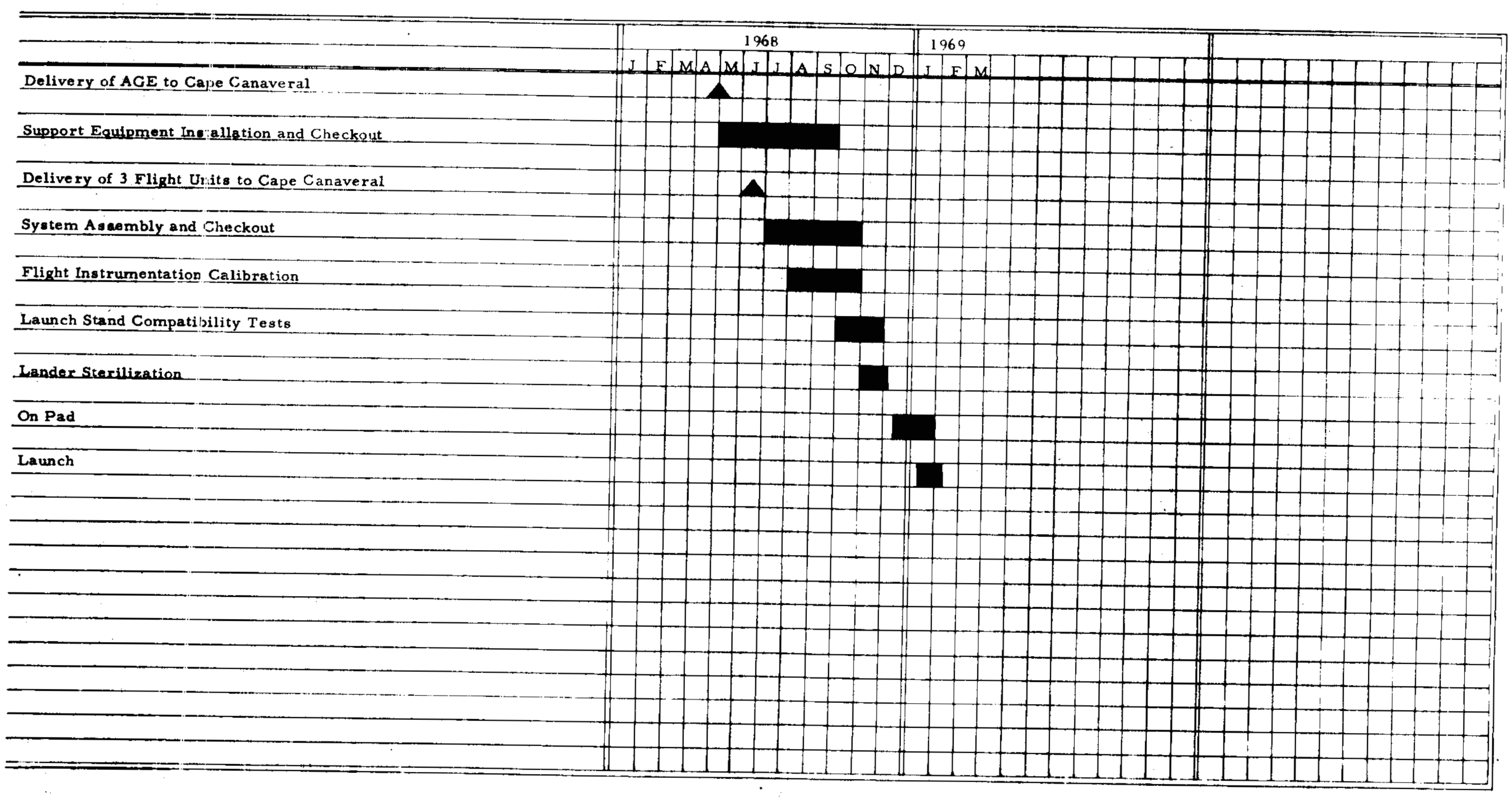
Most of the subsystems for the flight units will be available in the second and third quarters of 1967 for subassembly and functional tests in the last quarter of 1967. The orbiter and lander vehicles, propulsion systems, and the lander power supply system are delivered in the last quarter of 1967 for subassembly and functional tests. The flight unit scientific instruments are scheduled for delivery in the first quarter of 1968. Final assembly and acceptance testing of three flight units of both orbiter and lander take place in the first and second quarters of 1968. Sterilization of the lander occurs at the same time. Delivery of the three flight units is scheduled for mid-1968. It should be noted that the qualification test period immediately precedes the flight unit assembly and acceptance test period with no overlap. It should also be noted that 5 months is provided for the qualification testing, and this is regarded as a minimum schedule because allowance must be made for qualification failures and recycling of subsequent modifications to both qualification and flight units.

Delivery of the flight units in mid-1968 will provide 7 months for launch site checkouts prior to the 30-day launch window in January 1969. As shown in the launch site schedule (table 9), the AGE is delivered 2 months earlier than the flight unit delivery, and 5 months is provided for its installation and checkout. Schedules for system assembly and checkout, instrumentation calibration, launch stand compatibility tests, and lander sterilization are also indicated.

Several long-lead development items will require an early development cycle. In some cases, development programs should be initiated before the hardware contract go-ahead. These items are the sterilization facility, the radioisotope thermoelectric generator power supply for the lander, the orbiter propulsion system, and the scientific instrumentation. The sterilization facility development will require about 10 months for pilot-plant fabrication and testing and 20 months to build, staff, and test the full-scale facility. For this facility to be ready in time for the qualification testing program and subsequent acceptance testing of flight units, it is necessary that the pilot-plant program be initiated almost immediately (January 1964). This will facilitate completion of the full-scale facility in the last quarter of 1966. It is conceivable that both the pilot-plant program and full-scale facility will be developed under the Mariner B program and, therefore, eliminated as a long-lead item in the Voyager Program. The RTG with a radioactive fuel element will require early development by the AEC, and this item is also scheduled to begin in January 1964. Eighteen months is provided for delivery of developmental test prototypes, with 9 months scheduled

TABLE 9

LAUNCH SITE SCHEDULE FOR 1969 MARS LAUNCH



for development testing. The conceptual design of the propulsion system indicated the need to develop a new engine, which will be a long-lead item. Subcontract initiation is scheduled for 1 month after the prime contract go-ahead. This will facilitate delivery of qualification test units in the first quarter of 1967 in time for the qualification test program. Development of the scientific instrumentation for remote operation in a new and ill defined environment will require extensive study and testing and should begin immediately.

4. COSTS

The Voyager Program cost estimate has been subdivided in several ways to accommodate various cost analyses and program revisions which may be desired. The total program costs have been subdivided by fiscal years, subsystems and engineering disciplines, and by cost elements within each subsystem and engineering discipline. Both operational hardware costs and development costs are subdivided by launch window. Operational hardware costs include all acceptance test and field support costs associated with the spacecraft.

The costs indicated correspond to the launch profile indicated in table 10. Two spacecraft of the indicated configuration will be flown in each launch window. A completely assembled spare vehicle and a complete set of subsystem spares will also be available for each launch window. This spares complement is compatible with the sterilization philosophy presented.

With a terminal sterilization facility located at the field site, the maintenance of the lander requires a significantly smaller complement of subsystem spares than is required if the lander must be returned to the factory for maintenance. This difference in spares allotment is more costly than construction and operation of the field site sterilization facility.

The less ambitious launch program identified in table 10, which eliminates the Venus 1972 and 1975 launches, allows a reduction in total program cost from \$798.6 to \$668.5 million.

The influence of the Martian low-density atmospheric model on the program cost has not been evaluated in detail; however, some elements can be identified. If the lander configuration does not change, except in its external heat shield and structurel to reduce the M/C_DA, the lander per unit cost will not appreciably change. The larger lander size will not allow the orbiter/two lander or Bus/two lander configurations shown for the Mars 1971 and Mars 1975 launch opportunities, respectively. These opportunities will only accommodate an orbiter/lander configuration. This change in the proposed launch profile will reduce the total program cost by \$9 million.

Reducing the length of the development program by 9 months, delaying the program start date from October 1964 to July 1965, as shown in table 7, will increase the depvelopment costs if the same level of design effort is to be applied. A few of the subsystem design and development cycles must be compressed, requiring the application of overtime, and in some cases, duplicate developments will be necessary to ensure availability of flight-qualified hardware in sufficient time to accommodate the 1969 Mars flight. The design, development, test, qualification, and manufacturing cycles will overlap more severely, resulting in more costly changes to the spacecraft design late in the manufacturing process.

It is difficult to attach a specific cost to this schedule compression; however, past experience with crash development programs indicates a 10 to 20 percent increase in the development costs is associated with this degree of schedule compression.

TABLE 10

LAUNCH PROFILE

Mars 1969	O/L
Mars 1971	O/2L
Mars 1973	O/L
Mars 1975	B/2L
Venus 1970	O/3C
Venus 1972	O/3C*
Venus 1973	O/L
Venus 1975	O/L*

*Launch opportunity omitted in less ambitious launch profile

O - Oribter
B - Flybar Bus
L - Direct Entry Lander
C - Orbital Engry Capsule

A summary of the Voyager mission cost in presented in table 11.

TABLE 11

VOYAGER COST SUMMARY

	Direct Labor	Overhead 118%	Material	Spec. Test Equipment	Sub Contract	Material Handling 8%	Travel	Consultants	Computer	Subtotal	G & A 9%	Subtotal	Fee 8%	Subtotal	Facilities	Total
Spacecraft Design	$14,523,278	$17,137,469	$ 1,483,000	$ 450,000	$	$ 154,640	$ 260,500	$	$ 60,775	$ 34,069,662	$ 3,066,270	$ 37,135,932	$ 2,970,875	$ 40,106,807		
Structural Development	4,407,897	5,201,318	1,000,000	350,000		108,000	427,500	60,000	48,750	11,603,465	1,044,312	12,647,777	1,011,822	13,659,599		
Aerodynamics	7,415,000	3,749,700	324,000	200,000	2,140,000	213,120	198,000	50,000	1,491,750	20,781,570	1,870,341	22,651,911	1,812,153	24,464,064		
Heat Shield	4,722,415	5,572,450	2,052,000	1,225,000		262,160	360,000	42,000	120,250	14,356,275	1,292,065	15,648,340	1,251,867	16,900,207		
Thermal Control Subsystem Development	5,415,000	6,389,700	568,000	200,000		61,440	58,000	15,000	305,500	13,012,640	1,171,138	14,183,778	1,134,702	15,318,480		
Communication and Power Subsystem Development	7,054,451	8,324,252	5,299,000	450,000	11,568,092	1,385,367	270,500	122,500	22,750	34,496,912	3,104,722	37,601,634	3,008,131	40,609,765		
Mapping Subsystem Development	2,771,065	3,269,857	1,600,000		1,450,000	244,000	51,000	20,000	190,125	9,596,047	863,644	10,459,691	836,775	11,296,466		
Descent and Re-erection Subsystem Development	2,350,000	2,773,000			4,583,400	366,672				10,073,072	906,576	10,979,648	878,372	11,858,020		
Propulsion Subsystem	4,127,760	4,870,756	113,000		28,700,000	2,305,040				40,116,556	3,610,490	43,727,046	3,498,164	47,225,210		
Guidance and Stabilization Control System Development	7,428,676	8,765,838	511,000	4,700,000	4,930,000	811,280	190,500	13,800	200,038	27,551,132	2,479,604	30,030,736	2,402,458	32,433,194		
Aerospace Ground Support Equipment Development	2,978,656	3,514,814	1,971,750			157,740	71,500		32,500	8,726,960	785,426	9,512,386	760,991	10,273,377		
Test Support Program	17,918,554	10,637,648*					238,000			28,794,202	2,591,478	31,385,680	2,510,854	33,896,534		
Sterilization Program	3,615,613	4,266,423	419,000			33,520				8,334,556	750,110	9,084,666	726,773	9,811,439		
Earth Entry Test Program	1,170,000	1,380,600								2,550,600	229,554	2,780,154	222,412	3,002,566		
Drop Test Program	780,000	920,400								1,700,400	153,036	1,853,436	148,275	2,001,711		
Quality Test Program	195,000	230,100		4,300,000		344,000				5,069,100	456,219	5,525,319	442,026	5,967,345		
Systems Analysis and Integration	4,057,597	4,787,965					118,000			8,963,562	806,720	9,770,282	781,623	10,551,905		
Program Management	2,358,720	2,783,290					120,000			5,262,010	473,581	5,735,591	458,847	6,194,438		
Reliability and Quality Assurance	20,226,185	23,866,898			13,896,500	1,111,720	1,869,056	97,000	726,375	61,793,734	5,561,436	67,355,170	5,388,414	72,743,584		
Manufacturing and Quality Control	57,588,140	67,954,005	17,522,360		141,410,515	12,714,630	4,391,276	425,000	1,114,750	303,120,676	27,280,860	330,401,536	26,432,123	356,833,659		
Acceptance Tests	5,504,587	6,495,413								12,000,000	1,080,000	13,080,000	1,046,400	14,126,400		
Documentation	1,500,000	1,770,000								3,270,000	294,300	3,564,300	285,144	3,849,444		
Total	$178,108,594	$199,661,896	$32,863,110	$11,875,000	$208,678,507*	$20,273,329	$ 8,623,832	$ 845,300	$ 4,313,563	$665,243,131	$59,871,882	$725,115,013	$58,009,201	$783,124,214	$15,485,000	$798,609,214

*52% Overhead on $15,918,554 Offsite Direct Labor

TABLE 11 (Concl'd)

TOTAL COST (IN MILLION $)

F/Y	65	66	67	68	69	70	71	72	73	74	75	76	77	Total
Funding	68.6	124.4	105.3	74.2	67.8	57.4	60.6	55.0	59.8	50.2	48.6	26.1	.6	798.6

COST BY WINDOW (LESS DEVELOPMENT)

Development	M 69 O/L	M 71 O/2L	M 73 O/L	M 75 B/2L	V 70 O/3L	V 72 O/3L	V 73 O/L	V 75 O/L		Total
540.0	30.8	36.3	30.8	34.1	32.1	32.1	31.2	31.2		798.6

COST BY WINDOW (INCL. DEVELOPMENT)

M 69	M 71	M 73	M 75	V 70	V 72	V 73	V 75		Total
377.1	56.6	33.4	33.4	167.2	33.4	64.1	33.4		798.6

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